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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

NACA - INDUSTRY CONFERENCE ON PERSONAL AIRCRAFT RESEARCH

A COMPILATION OF THE PAPERS PRESENTED BY NACA STAFF MEMBERS

Langley Memorial Aeronautical Laboratory
Langley Field, Va.

September 20, 1946

NACA

NACA - INDUSTRY CONFERENCE

ON PERSONAL-AIRCRAFT RESEARCH

A Compilation of the Papers Presented
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The authors are members of the staffs of the NACA Laboratories
at Langley Field, Va., Cleveland, Ohio, and Moffet Field, Calif.
and the NACA Washington Office.

INTRODUCTION

This conference was organized by the NACA to acquaint airplane designers with the results of wartime research considered to be of interest in the design of personal aircraft and to discuss current and proposed NACA research projects applicable to light airplanes.

The technical papers presented at the conference were necessarily brief and are reproduced herein in the same form as they were given in the interest of prompt distribution. The original presentation and this compilation are considered as complementary to, rather than substitutes for, the Committee's complete and formal reports. A number of reports used as references herein, which were originally issued in classified form, are now declassified and are being printed and distributed as NACA Wartime Reports.

A list of the conferees is included.

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LIST OF CONFEREES

The following were registered at the NACA-Industry Conference on Personal Aircraft Research, Langley Memorial Aeronautical Laboratory, Langley Field, Va., September 20, 1946.

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Wright, T.P.	CAA

FLYING QUALITIES

HISTORY AND SIGNIFICANCE OF MEASURED FLYING QUALITIES

By Melvin N. Gough

The flying qualities of an airplane may be defined as the stability and control characteristics that have an important bearing on the safety of flight and on the pilot's impressions of the ease of flying and maneuvering an airplane.

The endeaver to develop a systematic quantitative measurement of flying characteristics has been entered into enthusiastically by producers, engineers, operators, and pilots alike. All have appreciated the necessity to evaluate the hitherto unsatisfactory relative opinions of pilots regarding their flight experiences. Pilots themselves desire a more factual method of expression; and engineers require quantitative information for knowledge of existing designs and future developments.

In the past, interest in this direction was also evidenced by the Guggenheim Safe Airplane Competition, the CAA safety program, the work of the Army and Navy testing centers, and the research programs of the NACA. There have been many contributors and much logical development. The increased need for recording instrumentation was apparent as more specific measurements of control movements and resulting airplane motions were required. The NACA continued to apply its facilities to the development of the necessary instrumentation and the accumulation of specific data, and at the beginning of the war had assembled complete quantitative information on twelve airplanes, including five of the private-owner or light-airplane type. From the fund of information accumulated in these tests, it was possible, early in 1941, for the NACA to prepare a set of requirements for satisfactory flying qualities in terms of quantities that had been measured in flight and could be calculated by engineers and could even be predicted from wind-tunnel tests.

When this country entered the war, all NACA facilities were directed to the immediate war effort. The Army and Navy revised the general NACA Flying Qualities Specifications to their immediate specific requirements, and looked to the NACA to continue its investigations on existing and new military aircraft. Manufacturers of military designs were familiarized with the information then available. By the end of the war the total number of airplanes tested was increased to 44 at the Langley Laboratory and to 16 at the new Ames Laboratory.

This vast experience is now available to all. The general NACA specifications were continually modified and expanded as necessary in the light of new developments and are now in the form of an NACA report (reference 1). It and many other reports of specific measurements made on airplanes are to be released for general use. The NACA requirements, as they now exist, are to be considered as a very minimum, which, when met, give reasonable assurance that the airplane will be safe to fly and desirable from the pilot's standpoint. For a light airplane, which may be flown by unskilled pilots and which must meet unusually high standards of flight safety, it will be desirable to obtain characteristics superior to those specified.

It is of interest to note that not only does there now exist a better understanding of why an airplane handles as it does and what is believed desirable, but a new terminology and a systematic approach understandable to both pilot and engineer has been developed. Even in the absence of quantitative measurements, the relative opinions of pilots familiar with the requirements are of greater value.

Throughout the day, other speakers will discuss some of the specific engineering aspects of the existing knowledge of flying qualities.

REFERENCE

1. Gilruth, R. R.: Requirements for Satisfactory Flying Qualities of Airplanes. NACA ACR, April 1941.

FLYING QUALITIES REQUIREMENTS FOR PERSONAL AIRPLANES

By William H. Phillips

A great deal of research work has been conducted during the war years to determine requirements for satisfactory flying qualities of airplanes. While previously there was considerable speculation as to the flying characteristics desired in an airplane, it is now possible to specify in quantitative terms the minimum requirements that an airplane must meet in order to be considered satisfactory from the pilot's standpoint. These requirements were set up as a result of experience gained in testing all types of airplanes, including five light airplanes. The requirements may therefore be used with confidence by the designer of personal airplanes to arrive at a design with satisfactory stability and control characteristics.

The requirements may be listed under the general headings of longitudinal stability and control characteristics, lateral stability and control characteristics, and stalling characteristics. Table I presents a list of the factors considered in the requirements for longitudinal stability and control characteristics. I would like to point out how NACA research has contributed to the knowledge of some of these topics.

First are listed the requirements for the elevator control in take-off. Tests of numerous wind-tunnel models have been made with a ground-board in place in the tunnel to determine the ability of airplanes of various designs to meet this requirement.

Next are the requirements for elevator control in steady flight. As a result of an analysis that has been made of the flight tests of numerous airplanes (reference 1), it is now possible to predict from the airplane's dimensions the power-off static longitudinal stability of an airplane, and hence the basic elevator control characteristics in steady flight, with the same degree of accuracy that it may be measured in flight or in the wind tunnel. In addition, a great deal of data have been accumulated on the effects of power on stability, both from flight and wind-tunnel tests. Flight-test data are summarized in reference 2. The control forces required to fly steadily at various speeds may also be predicted as a result of extensive NACA research on control-surface hinge moments and aerodynamic balances. (For example, see references 3 and 4.)

The requirements for the longitudinal trimming device are next on the list. A report is available which summarizes the results of many wind-tunnel and flight tests on the effectiveness of trimming tabs. This report (reference 5) will allow an accurate estimation of the tab sizes to meet this requirement.

The elevator control characteristics in accelerated flight have been found to be very important in determining the pilot's opinion of an airplane. Quantitative limits for the control motions and force gradients necessary on airplanes of many different types have been determined from flight measurements. A new airplane may be designed to incorporate these characteristics by the methods described previously.

The characteristics of the uncontrolled longitudinal motion are considered next. It may be of interest to mention in passing that the characteristics of the long-period, or phugoid oscillation of an airplane have been found to be relatively unimportant. The extensive theoretical work conducted on this subject in the past therefore has little bearing on the subject of flying qualities. Sometimes the short-period-oscillation characteristics of an airplane may fail to meet the requirements, however. Recent reports are available which describe flight experiences with unsatisfactory characteristics (reference 6), as well as theoretical analyses to show how to avoid the undesirable characteristics (reference 7).

Trim changes due to power and flaps are limited to definite quantitative values in the requirements. The light airplane manufacturer should be encouraged to note that in the case of several fighter-type airplanes, where the problems of reducing the control force changes due to flaps and power are much more difficult than for a smaller airplane, designs with very small trim changes have been developed as a result of wind-tunnel and flight tests. The problem of reducing the trim changes in the case of a light airplane should therefore be fairly easy.

The elevator control characteristics in landing are specified in the final longitudinal requirement. A report is available which presents a method for calculating the elevator angle required to land (reference 8). This method has been developed from an analysis of the results of flight tests of numerous airplanes. Extensive wind-tunnel data are also available on this subject. The elevator angle required to land is of importance because this is frequently the most critical requirement for elevator effectiveness.

Table I also shows the requirements for lateral stability and control. The requirements for aileron control characteristics are expressed quantitatively in terms of the minimum value of the helix angle generated by the wing tip in a roll. This requirement was set up before the war, and numerous subsequent tests have confirmed its validity. A summary report on NACA lateral-control research has been prepared (reference 9). This report contains sufficient data to allow the design of satisfactory ailerons for any type of airplane.

The yaw due to ailerons should not exceed a certain maximum value. This requirement has been found to be a critical one from the standpoint of directional stability. Ordinarily changing the aileron design does not greatly reduce the adverse yawing moments at high lift coefficients.

The rudder and aileron trimming devices may be designed to meet the requirements in the same manner as the longitudinal trimming devices.

The next requirements specify the limits of rolling moment due to sideslip, ordinarily known as dihedral effect. Several flight investigations made with airplanes of various plan forms and different amounts of geometric dihedral have shown the allowable limits of dihedral effect for satisfactory flying qualities.

The rudder control characteristics are considered next in the list of requirements. The rudder must perform many functions besides simply providing directional trim in the various flight conditions. These functions include offsetting the adverse aileron yawing moments, providing satisfactory control in sideslips, providing adequate spin recovery, providing satisfactory control during take-off and landing, and offsetting the yawing moment due to asymetric power on a multiengine airplane. A great deal of research has been done to determine the rudder configurations required to satisfy these requirements and in later lectures the design criterion for spin recovery will be discussed in more detail. It has been shown by flight tests that the rudder effectiveness required for take-off and landing is determined by many factors other than the design of the rudder.

The yawing moment due to sideslip or directional stability must be sufficient to meet certain requirements for providing satisfactory sideslip characteristics, limiting the sideslip in rolls due to aileron yawing moments, and providing adequate stability for flight with asymetric power. A large amount of directional stability has never failed to be beneficial to the handling characteristics of an airplane. Design data are available to allow estimation of the tail size required to provide adequate directional stability, as will be discussed in a later paper. The cross-wind force characteristics and the pitching moment due to sideslip, both of which are characteristics measured in steady sideslips, form the subject of the next requirement. The pitching moment due to sideslip should be small as this factor may lead to inadvertent stalling. Typical flight measurements of satisfactory sideslip characteristics are shown in figure 1. The final requirement for lateral stability and control is concerned with the uncontrolled lateral and directional motion. The results of flight tests have been used to establish these requirements and a great deal of theoretical work performed by the NACA is available to enable the designer to predict these characteristics (reference 10).

Time does not permit a detailed discussion of all the requirements. The characteristics of a personal airplane should, in general, be superior to the minimum requirements for satisfactory flying qualities. Some of the characteristics that are believed very desirable in a light airplane as a result of NACA research are as follows: the static longitudinal stability in all flight conditions should be large, so that the pilot will be warned of the approach to the stall by the rearward position of the stick at low flight speeds. The stick-force gradients in straight flight should be stable and should be sufficiently large compared to the control friction so that the stick will have a definite centering tendency. The force variation with acceleration in accelerated flight should be between 7 and 10 pounds per g. The trim changes due to changing flap or power condition should be very small.

The aileron control effectiveness should be sufficient to provide a helix angle of the wing tip of greater than 0.07 radian. The aileron forces should be light, but sufficiently large compared to the friction to give the control stick a definite centering tendency. The directional stability should be large. The rudder control should be sufficiently powerful to overcome aileron yaw.

The stalling characteristics of a personal-type airplane are very important. The stall should be preceded by adequate warning in the form of buffeting of the airplane and by a marked increase in rearward motion in the stick and pull force as the stall is approached. When the airplane is stalled, there should be no rolling instability in any flight condition, even with the stick held full back. This characteristic was obtained by relatively minor modifications to a typical light airplane at the NACA (reference 11).

In conclusion it may be pointed out that a large amount of information has been accumulated both on the requirements for satisfactory flying qualities and on the means for designing an airplane to incorporate these flying qualities. As a result, it is believed possible to design an airplane of the personal-airplane category with the assurance that its flying qualities will be satisfactory.

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9 Research Department (Compiled by Thomas A. Toll): Summary of Lateral-Control Research. (Prospective TN.)

10. Greenberg, Harry, and Sternfield, Leonard: A Theoretical Investigation of the Lateral Oscillations of an Airplane with Free Rudder with Special Reference to the Effect of Friction. NACA ARR, March 1943.

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Table I - OUTLINE OF FLYING QUALITIES REQUIREMENTS

1. Requirements for Longitudinal Stability and Control:

   A. Elevator control in takeoff

   B. Elevator control in steady flight

   C. Longitudinal trimming device

   D. Elevator control in accelerated flight

   E. Uncontrolled longitudinal motion

   F. Limits of trim change due to power and flaps

   G. Elevator control in landing

11. Requirements for Lateral Stability and Control

   A. Aileron-control characteristics

   B. Yaw due to ailerons

   C. Rudder and aileron trimming devices

   D. Limits of rolling moment due to sideslip

   E. Rudder-control characteristics

   F. Yawing moment due to sideslip

   G. Cross-wind force characteristics

   H. Pitching moment due to sideslip

   I. Uncontrolled lateral and directional motion

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111. Stalling Characteristics

Phillips

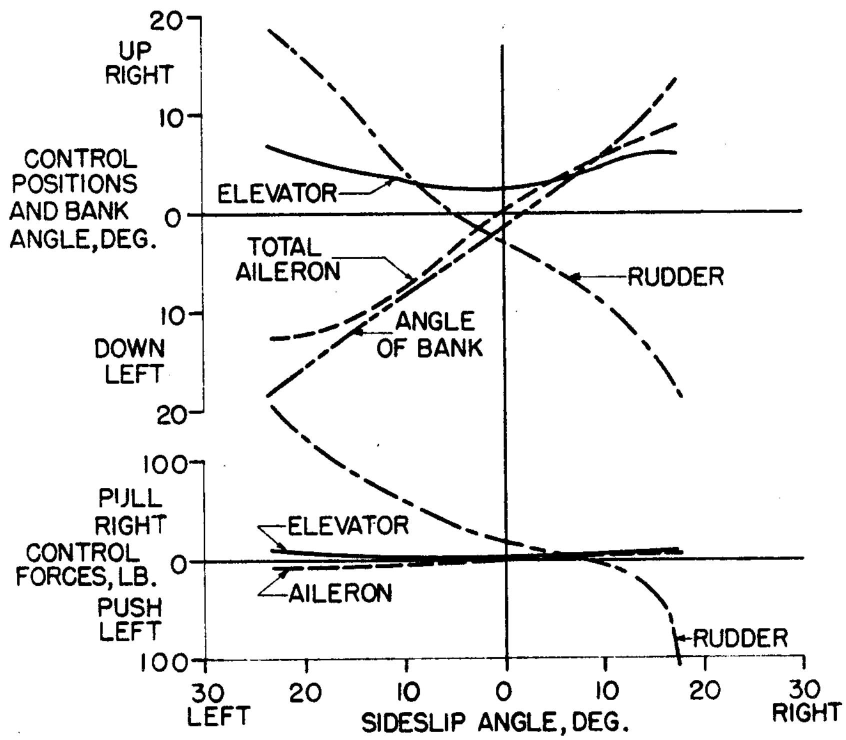

Figure 1.- Steady sideslip characteristics.

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STABILITY AND CONTROL

EFFECTS OF INDIVIDUAL STABILITY FACTORS ON FLYING CHARACTERISTICS OF AIRPLANES

By Marion O. McKinney, Jr.

The previous paper by Mr. Phillips presented a discussion of the flying qualities that an airplane should have in order to be safe and easy to fly. For the sake of convenience these flying qualities requirements may be divided into three groups - those concerned with the stability of the airplane, those concerned with the control characteristics of the airplane, and those concerned with stalling and spinning. The present paper will discuss the effects of the individual stability factors on flying qualities concerned with the stability of the airplane. The proper design of the controls and considerations of stalling and spinning will be discussed in some of the following papers.

The NACA has conducted some research during the war to determine the effects of many of the basic stability factors on flying qualities. Much of this work has been done in the free-flight tunnel where it has been found convenient and economical to vary the basic stability factors independently of each other. Such variations of the individual stability parameters are accomplished by varying the center-of-gravity position, horizontal or vertical tail size or tail length, and the dihedral angle.

In the conventional manner the longitudinal flying qualities will be treated first and the lateral flying qualities will be treated second.

For many years the longitudinal stability has been considered to be primarily dependent upon the center-of-gravity location and the damping in pitch or horizontal tail size and tail length. The results of an investigation in the free-flight tunnel to determine the effects of these two factors on the longitudinal flying qualities (reference 1) are presented in figure 1. Here the pilot's opinions of the longitudinal steadiness (as indicated by the qualitative ratings: good, fair, poor, and dangerous) are shown as functions of the center-of-gravity location and damping in pitch. The center-of-gravity location is presented in terms of the static margin which is the distance in chords that the center of gravity is ahead of the neutral point or aerodynamic center of the complete airplane. The damping in pitch is presented in terms of the stability derivative C_{m_q} which is the rate of change of the pitching-moment coefficient with a pitching-velocity factor $\left(\partial C_m/\partial \frac{qc}{2V}\right)$.

The range of values of C_{m_q} covered on this chart represent a range of airplane configurations from the straight-wing tailless type to one having a horizontal tail 24 percent of the wing area 2 chords aft of the center of gravity.

It is apparent that the pilot of the model considered that the longitudinal steadiness is virtually independent of the damping in pitch and is almost entirely dependent upon the center-of-gravity position. The chart indicates that if the center of gravity is more than 0.08 chord ahead of the aerodynamic center of the airplane that the longitudinal steadiness will be good. This conclusion has been supported by full-scale flight tests.

Similar investigations have been made to determine the effects of various basic lateral stability factors on the lateral flying qualities. The results of an investigation in the free-flight tunnel to determine the effects of dihedral angle and vertical-tail area on the lateral flying qualities are presented in figure 2. This chart was prepared from data presented in reference 2. The pilot's opinion of the lateral flight behavior is presented as functions of the directional stability and effective dihedral. The directional stability is expressed in terms of the nondimensional stability derivative C_{n_β} $\left(\frac{\partial C_n}{\partial \beta}\right)$ which is the rate of change of the yawing-moment coefficient with angle of sideslip, and the range of values of C_{n_β} covered on the chart represent vertical tails from approximately 0 to 30 percent of the wing area located one-half of the wing span aft of the center of gravity. The effective dihedral is expressed in terms of the stability factor C_{l_β} $\left(\frac{\partial C_l}{\partial \beta}\right)$ which is the rate of change of rolling-moment coefficient with angle of sideslip, and the range of values of C_{l_β} covered on the chart represent approximately effective dihedral angles from -10° to 20°.

It is apparent that the pilot thought that the model had the best lateral flight behavior when it had a small positive effective dihedral and high directional stability.

It was believed that this chart presented a picture of the most important problem in obtaining good lateral flying qualities because these two parameters (dihedral effect and directional stability) are the primary factors affecting lateral stability. Investigations were made, however, to determine how variations of certain other lateral stability parameters would affect the flight behavior boundaries presented in figure 2. The factors investigated were wing loading, mass distribution, and the aerodynamic parameters concerned with the lateral area and damping in yaw of an airplane. The results of these investigations are presented in references 3 to 6. In brief, these investigations showed that there was virtually no effect of reasonable changes in these factors on the flight behavior boundaries of figure 2.

Although the flying qualities requirements do not specify that an airplane should be spirally stable, it is realized that spiral stability might be a desirable feature for a personal owner airplane so that it might be completely stable and capable of flying itself to the greatest possible extent. The spiral stability problem of light airplanes was therefore analyzed. It was found that it was not possible to have spiral stability over the entire speed range and still have good lateral flight behavior. It is possible, however, to have spiral stability in the high-speed and cruising conditions and still have good lateral flight behavior if the airplane were so designed that it would fall near the right-hand branch of the good flight behavior boundary of figure 2.

From these considerations it may be concluded that if the minimum satisfactory size vertical tail is to be used an airplane should have about 5° effective dihedral $\left(C_{l_\beta} = -0.001\right)$ and directional stability corresponding to a value of C_{n_β} of 0.002 which represents a somewhat larger tail than is generally used on current light airplanes.

Figure 2 has been correlated with the results of flying qualities investigations of quite a few full-scale airplanes and has been found to be in good agreement with the full-scale flight tests. That is, if the effective dihedral and directional stability of an airplane would place it in the good flight behavior region of this chart, it will have good lateral flying qualities.

In order to illustrate the effects of increasing the dihedral angle we may refer to some flight tests made by the NACA on a light airplane several years ago (reference 7). The four symbols on the chart (fig. 2) represent the locations of that airplane on the chart as the dihedral angle was varied from 0° to 3° to 6° and 9°. With the smallest dihedral angle the airplane was found to fly the best. As the dihedral angle was increased, the flying characteristics became worse. With 9° dihedral the flying characteristics were quite poor, primarily because of the adverse yawing due to rolling and aileron deflection which caused rolling moments of about the same magnitude as these caused by the ailerons, thus causing the airplane to be very difficult to fly.

In order to illustrate the effects of changing the vertical tail area figure 3 was prepared. The calculated rolling and yawing velocities following an abrupt rolling moment (such as might be produced by a rolling gust or by an aileron giving a pure rolling moment) are shown as function of time for an airplane with two vertical tails - a small tail, $C_{n_\beta} = 0.00042$, and a larger tail, $C_{n_\beta} = 0.00168$. The airplane with the larger tail reaches its maximum rolling velocity in about 1 second and holds that rolling velocity fairly steadily. The airplane with the smaller tail, however, never reached a steady rolling state. The initial adverse yawing was greater for the airplane with the smaller tail, and the angular accelerations (as indicated by the slopes of these curves, $\frac{dp}{dt}$ and $\frac{dr}{dt}$) were greater for the airplane with the smaller tail. It may therefore be concluded that the airplane with the larger tail would have the better response to its controls and would ride smoother through a series of rolling gusts.

There are other factors which affect the flying qualities, such as the control and stalling characteristics which have not been considered in the present paper. It is necessary, however, that an airplane have certain basic stability characteristics in order to have satisfactory flying qualities. It seems, at present, that the problem of obtaining these basic stability characteristics can be reduced to a consideration of three factors. The center of gravity must be far enough ahead of the aerodynamic center of the airplane to obtain good longitudinal flying qualities, and the dihedral angle and vertical tail area must be properly proportioned to obtain good lateral flying qualities.

SYMBOLS

C_{m_q} rate of change of pitching-moment coefficient with pitching velocity factor $\left(\frac{\partial C_m}{\partial \frac{qc}{2V}}\right)$

C_{n_β}	rate of change of yawing-moment coefficient with angle of sideslip $\left(\frac{\partial C_n}{\partial \beta}\right)$
C_{l_β}	rate of change of rolling-moment coefficient with angle of sideslip $\left(\frac{\partial C_l}{\partial \beta}\right)$
C_m	pitching-moment coefficient $\left(\frac{M}{qcS}\right)$
C_n	yawing-moment coefficient $\left(\frac{N}{qbS}\right)$
C_l	rolling-moment coefficient $\left(\frac{L}{qbS}\right)$
C_L	lift coefficient $\left(\frac{Lift}{qS}\right)$
c	mean aerodynamic chord, feet
b	wing span, feet
S	wing area, square feet
M	pitching moment, foot-pounds
N	yawing moment with respect to stability axes, foot-pounds
L	rolling moment, foot-pounds
q	dynamic pressure, pounds per square foot
V	airspeed, feet per second
p	rolling velocity, radians per second
r	yawing velocity, radians per second
t	time, seconds
β	angle of sideslip, degrees
M	pitching moment
L	rolling-moment with respect to stability ax's, foot-pounds

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McKinney

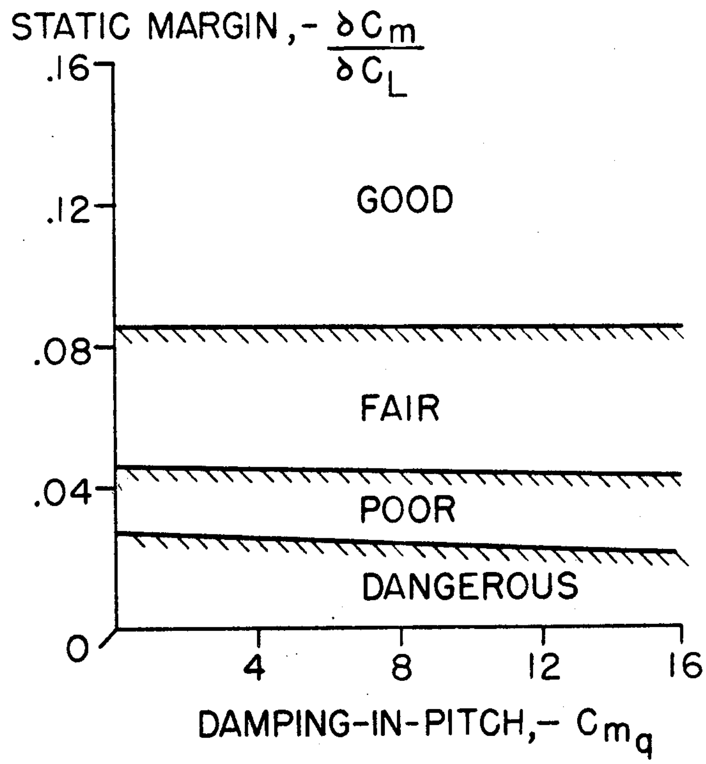

Figure 1.- Effect of center-of-gravity location and damping in pitch of longitudinal steadiness.

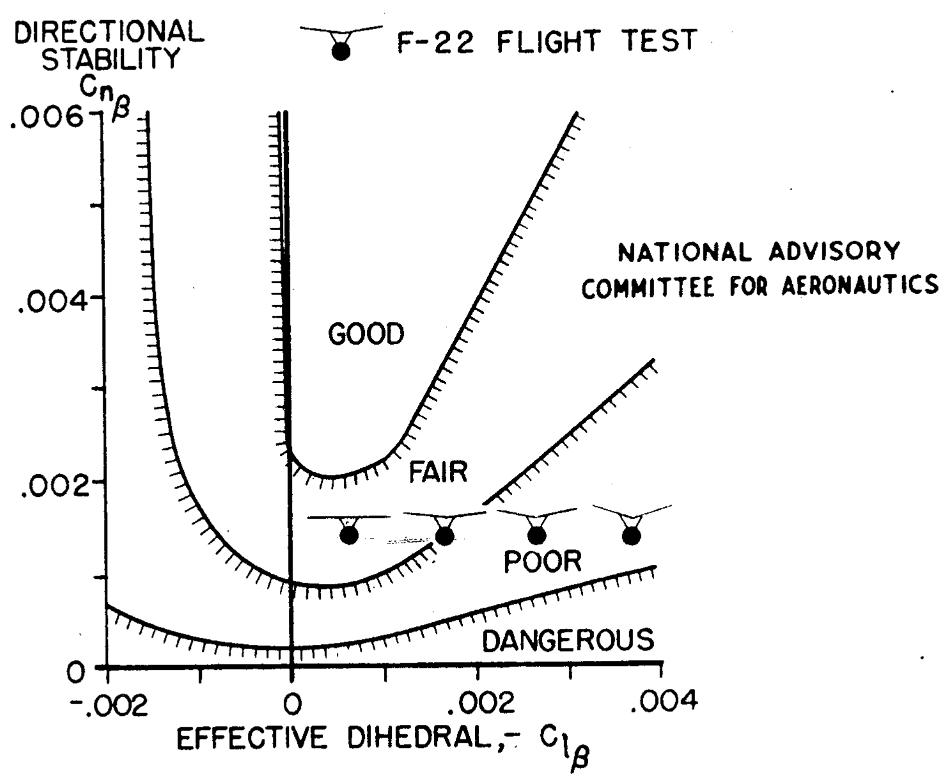

Figure 2.- Effect of vertical-tail area and dihedral on lateral flight behavior.

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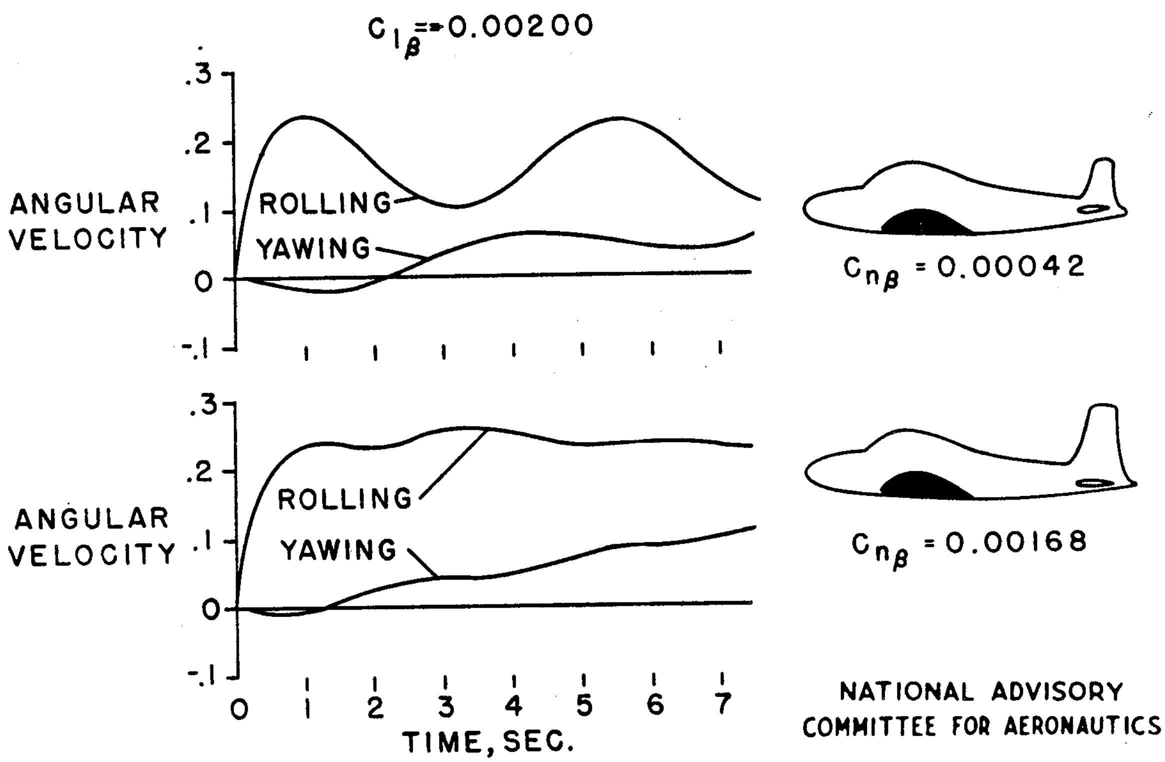

Figure 3.- Effect of directional stability on response to a rolling moment.

PROPORTIONING THE AIRPLANE FOR LATERAL STABILITY

By Charles J. Donlan

In proportioning the airplane for lateral stability, the two characteristics of most importance are the directional stability factor C_{n_β} defined as the slope of the curve of yawing-moment coefficient against sideslip, and the effective dihedral factor C_{l_β} defined as the slope of the curve of rolling-moment coefficient against sideslip. The advantages of combining high values of C_{n_β} and low values of C_{l_β} have been amply demonstrated. The design problem is to provide the optimum ratio of C_{n_β} to C_{l_β} and to determine how both of these parameters and their ratio are apt to change for the various flight conditions encountered. The purpose of this brief paper is to summarize the design factors that influence these parameters and to indicate how reliably they can be estimated.

The basic forces influencing the directional stability of an airplane are indicated in figure 1. The forces and moments contributing to the yawing moment about the center of gravity are: (1) the propeller side force N, (2) the basic fuselage yawing moment N_F, and (3) the side force on the vertical tail L_T. We have found that side force developed by the propeller becomes important for large propellers located at some distance from the center of gravity. In any event, however, the propeller side force presents no serious design problem inasmuch as the variation of the propeller side force with angle of sideslip can be predicted very reliably and rapidly from design charts such as those presented in a paper by Ribner. (See reference 1.)

The unstable yawing moment of the fuselage also can be estimated fairly reliably either from experimental results on similar shaped bodies or by theories such as Multhopp's. (See reference 2.)

The side force developed by the vertical tail and its variation with angle of sideslip however cannot be estimated as reliably or as easily. The contribution of the vertical tail to directional stability is proportional to the slope of the tail-normal-force curve, which, in turn, is primarily a function of the aspect ratio of the tail. It must be remembered, however, that the horizontal tail and fuselage act as an endplate which may cause the effective aspect ratio of the vertical tail to differ from its geometric value by 50 percent or more. Consequently, it is rather important to estimate how large the endplate effect may be. Methods for doing this are discussed extensively in papers by Pass (reference 3) and Murray (reference 4).

The effectiveness of the vertical tail is also influenced by the sidewash associated with the flow about the fuselage-wing combination. We have found that the sidewash is generally favorable for low-wing airplanes and adverse for high-wing airplanes. The sidewash effect is important and should be taken into account in estimating the directional stability. Methods of estimating this effect together with the complications introduced by flaps and power are discussed extensively in several of the papers now available to you.

In addition to supplying adequate stability at small angles of sideslip, we have frequently found it necessary to increase the effectiveness of the vertical tail at large angles of sideslip. One device commonly employed for this purpose is the dorsal fin. The typical effects produced by a dorsal fin are illustrated in figure 2. In this figure the yawing-moment coefficient C_n is plotted as a function of the sideslip angle β for an airplane with a vertical tail used in conjunction with a dorsal fin. You will notice that whereas the vertical tail without the dorsal fin becomes ineffective at an angle of sideslip of about 15°, the vertical tail in combination with the dorsal fin retains its effectiveness out to 30° without any detrimental effects at small angles of sideslip. The asymmetry of the curves is caused by the propeller slipstream and is typical of the effects introduced by power.

The dihedral effect exhibited by airplanes is, of course, closely associated with wing position, but fairly reliable estimates can be made of the effect. In general, high-wing arrangements exhibit greater dihedral effect than low-wing arrangements. The magnitude of the interference effect is illustrated in figure 3. These results were taken from reference 5 and are for an aspect-ratio 6 wing with no geometric dihedral. The ordinate on the left is the effective dihedral parameter C_{l_β} and the equivalent geometric dihedral angle is given on the right. The relative wing position is represented along the horizontal axis. It will be noted that the high-wing position is equivalent to about 5° of effective dihedral and the low-wing position about -5°. The fuselage cross-sectional shape appears to be of minor importance.

Any dihedral effect introduced by setting the wing at a given geometric dihedral angle can be added to the basic arrangement. We have found that with low-wing arrangements that the dihedral effect is reduced when power is applied, but the effect depends to a considerable extent on the amount of wing immersed in the slipstream. For high-powered airplanes, the power effect on dihedral can be extremely pronounced when flaps are immersed in the slipstream and many service types exhibit negative dihedral effect in the full-power, flap-down condition. For low-powered airplanes, the effect is probably unimportant.

SYMBOLS

C_{n_β} slope of curve of yawing-moment coefficient against sideslip

C_{l_β} slope of curve of rolling-moment coefficient against sideslip

C_n yawing-moment coefficient (N/qbS)

C_l rolling-moment coefficient (L/qbS)

N propeller side force, pounds: also, yawing moment with respect to stability Z axis, foot-pounds

L rolling-moment coefficient with respect to stability X axis, foot-pounds

N_F basic fuselage yawing moment, foot-pounds

L_T side force on vertical tail, pounds

Y total side force, pounds

T thrust (See fig. 1.)

L_W wing lift, pounds

b wing span, feet

S wing area, square feet

q dynamic pressure, pounds per square foot

β sideslip angle, degrees

ϕ angle of bank, degrees

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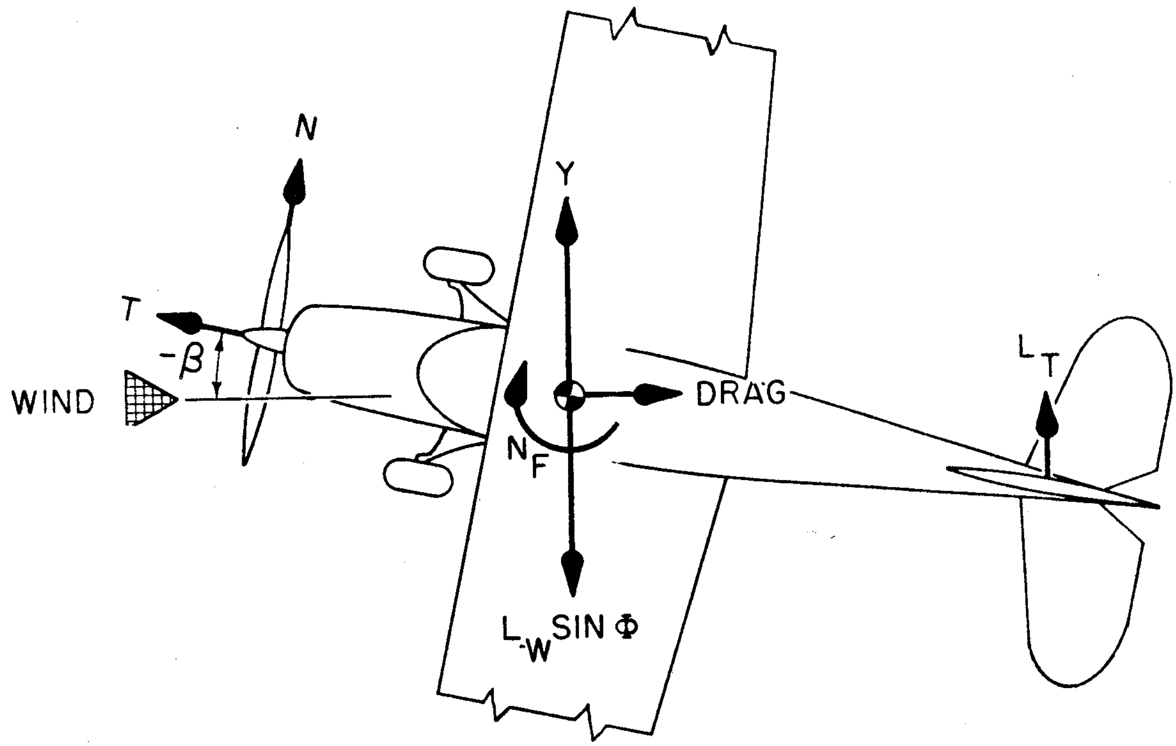

Figure 1.- Basic forces on an airplane in a sideslip.

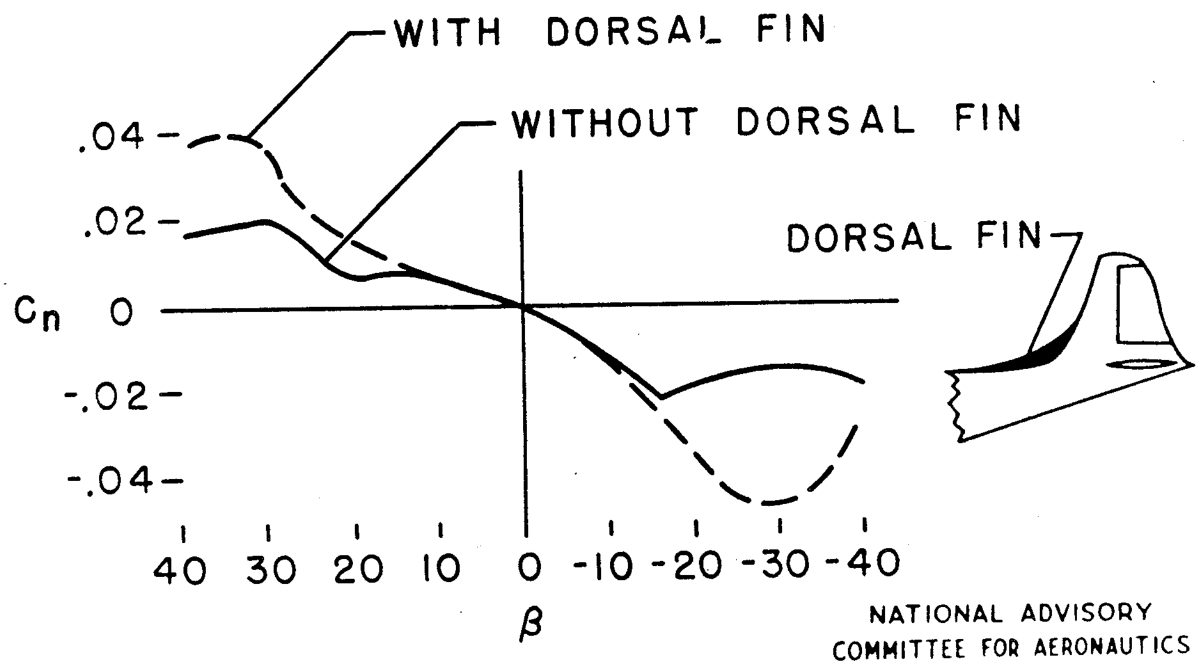

Figure 2.- Effect of dorsal fin on directional stability.

Donlan

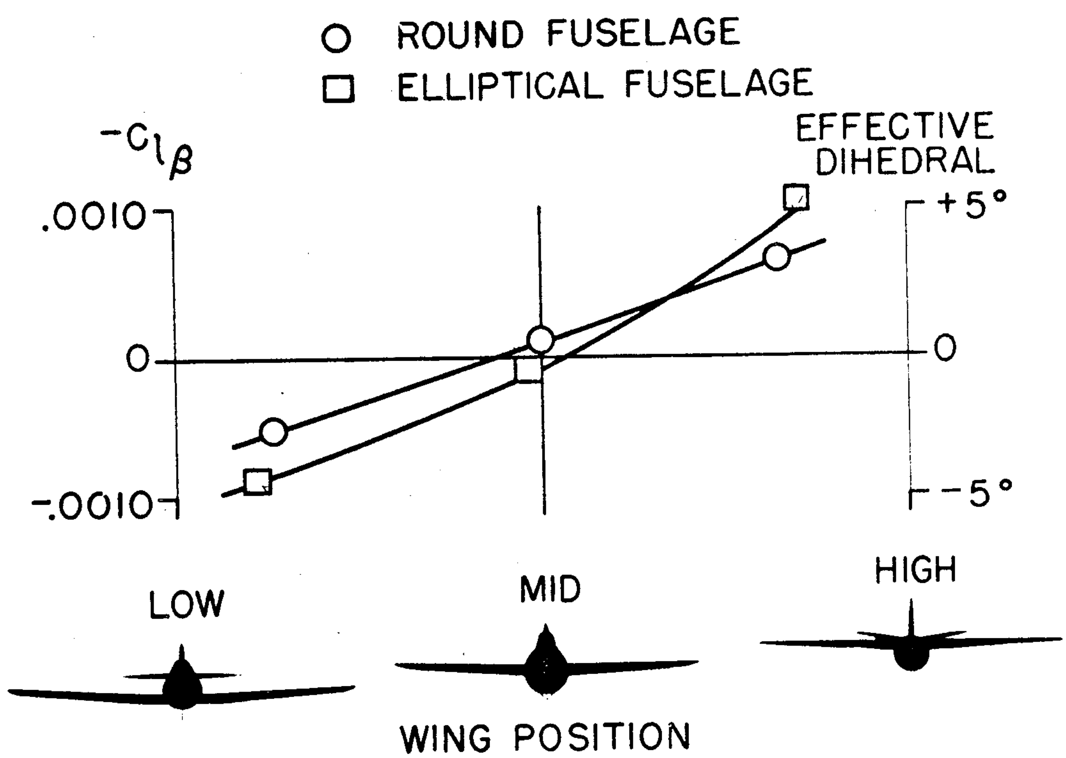

Figure 3.- Effect of wing position on dihedral effect.

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DESIGN OF CONTROL SURFACES

By Thomas A. Toll

From the year 1941 to the present time, a considerable amount of research related to control-surface design has been performed by the NACA laboratories. During these years we have been primarily interested in the development of control surfaces capable of providing high maneuverability at high speeds with reasonably low control forces; consequently, the problem of control-surface hinge moments has been at least as important as the problem of control-surface effectiveness. In the usual case, the effectiveness could be predicted quite accurately from information that was available in 1941, but the factors affecting control-surface hinge moments were not well understood. The greater part of recent research effort in control-surface design therefore has been directed toward the general problem of hinge moments and, in particular, toward methods of obtaining aerodynamic balance. A large part of the following discussion is concerned with the application of the results of recent research on hinge moments to the design of control surfaces for light aircraft.

Figure 1 shows hinge-moment characteristics of a typical control surface. Values of the hinge-moment coefficient are plotted against control-surface deflection for various fixed angles of attack. For most plain (or unbalanced) surfaces and for surfaces with well-designed balances, the variations of hinge-moment coefficient with angle of attack and with deflection are approximately linear over moderate ranges. Over these moderate ranges, therefore, the hinge-moment characteristics of a given control surface may be specified by two parameters - $\partial C_h/\partial \alpha$, which is the variation of hinge-moment coefficient with angle of attack and $\partial C_h/\partial \delta$, which is the variation of hinge-moment coefficient with deflection. Control characteristics for many flight conditions may be expressed mathematically in terms of these two parameters.

Figure 2 shows the relation of the hinge-moment parameters for certain elevator control characteristics for a typical airplane. In this figure, values of $\partial C_h/\partial \alpha$ are given on the ordinate scale and values of $\partial C_h/\partial \delta$ are given on the abscissa scale. The solid line is the boundary for static stability in level flight and is the condition for which there is no variation in stick force for changes in speed from the trim condition in level flight. Combinations of the hinge-moment parameters that would be represented by points in the region above this boundary would give a stable variation of stick force with speed, whereas combinations of the hinge-moment parameters that would be represented by points in the region below this boundary would give an unstable variation of stick force with speed. The dashed line is the boundary for maneuvering stability and is the condition for zero variation in stick force with changes in normal acceleration in a pull-up or a turn. The region above this boundary represents conditions for a stable variation of stick force with normal acceleration and the region below this boundary represents conditions for an unstable variation of stick force with normal acceleration. The slopes of these boundaries were calculated by the methods of reference 1 for a particular center-of-gravity location on a typical airplane. The slopes will vary with the different center-of-gravity locations that might exist. A point that represents a combination of hinge-moment-parameter values for a typical plain elevator is shown on this plot. The location of this point relative to the two boundaries indicates that such an elevator would provide stable stick forces either in level flight or in maneuvers. The amount of stability, however, is proportional to the distance from the boundaries to the point representing the elevator. For light aircraft, the stick forces should be highly stable and, therefore, it may be desirable to alter the hinge-moment parameters of the elevator in order to move the point representing the elevator farther from the boundaries.

The values of the hinge-moment parameters can be altered by the addition of aerodynamic balance. Most balances cause a decrease in the negative values of both of the hinge-moment parameters. It should be noted that a decrease in the negative value of $\partial C_h/\partial \alpha$ is favorable for elevator control provided that it is not accompained by too great a decrease in the negative value of $\partial C_h/\partial \delta$, since the object is to move the point farther from the boundaries. In the case of the rudder, it also seems desirable to decrease the negative value of $\partial C_h/\partial \alpha$ above the value for a plain rudder while still maintaining a high negative value of $\partial C_h/\partial \delta$. The value of $\partial C_h/\partial \alpha$ for the rudder never should be raised much above zero, however, because positive values of this parameter may cause snaking oscillations with controls free. (See references 2 and 3.)

Figure 3 shows how the values of the hinge-moment parameters of a control surface are affected by the addition of various kinds of aerodynamic balance. The characteristics of each of these balancing devices have been investigated extensively and are discussed in detail in reports that now are being released. (See references 4 to 7.)

Figure 3 shows that the addition of an unshielded horn balance (A) to a control surface causes a larger change in $\partial C_h/\partial \alpha$ than in $\partial C_h/\partial \delta$. A balancing tab (F) - that is, a tab geared to deflect in a direction opposite to the deflection of the control surface -

causes a negligible change in $\partial C_h/\partial\alpha$ for a given change in $\partial C_h/\partial\delta$. Other devices, such as the beveled trailing edge (B), the symmetrical overhang balances (C and D), and the sealed internal balance (E) have intermediate effects. The shielded horn balance and the Frise balance cannot be represented very satisfactorily by this type of chart because their characteristics are very nonlinear. Their effects on control forces however are similar to the effects of the symmetrical overhang balances.

From considerations of ease of manufacture and maintenance the unshielded horn balance (A), the symmetrical overhang balances (C and D), and the Frise balance probably are of greatest interest in light-aircraft design. The most desirable type of balance for a given control surface can be selected only after making a thorough study of the required functions of the control surface but, in general, for light aircraft, the unshielded horn balance is best suited to elevators or rudders and the Frise or symmetrical overhang balances are best suited to ailerons.

Figure 4 shows how the addition of aerodynamic balance to the elevator of a typical light airplane affects the stick force per unit normal acceleration - commonly called the stick force per g - over a range of center-of-gravity locations. For the plain elevator, the force per g varies from a high positive value to a small negative value as the center of gravity is moved from 20 percent of the mean aerodynamic chord to 40 percent of the mean aerodynamic chord. The addition of a round overhang balance makes the forces low throughout the center-of-gravity range, while the addition of an unshielded horn balance raises the forces through the greater part of the center-of-gravity range. The force per g can be raised by a constant increment throughout the center-of-gravity range by attaching a bobweight to the control stick. The effect of a bobweight is illustrated in figure 4 for the case of a plain elevator, but similar effects would be obtained through the use of a bobweight in conjunction with either of the balanced elevators. It is evident from these results that a considerable amount of control over the force per g is available through the use of aerodynamic balance or a bobweight.

The problem of aileron balance is influenced to a large extent by the dimensions of the ailerons and by the deflection range. Figure 5 shows combinations of span, chord, and deflection range of plain ailerons that should be capable of meeting the requirement $pb/2V = 0.07$ with no regard to control force. The aileron chord as a fraction of the wing chord is plotted against the aileron span as a fraction of the wing semispan. If a lift flap is to be used inboard of the ailerons, it may be desirable to restrict the ailerons to a small span. Figure 5 indicates that the requirement $pb/2V = 0.07$ can be met with a very small aileron span provided the aileron chord is of the order of 0.30 to 0.40 of the wing chord and the total deflection range is about 40°. Increases in the total deflection range to above 40° are of very little benefit because aileron effectiveness drops off rapidly at the higher aileron deflections. Although the $pb/2V$ requirement can be met with a small-span, wide-chord aileron, flight experience (reference 8) has not shown much promise for such ailerons, principally because they exhibit undesirable floating tendencies in sideslip. Careful design of the ailerons with regard to their hinge-moment characteristics may, however, provide some improvement in this respect.

Although it is possible to meet the requirement of a $pb/2V$ of 0.07 with widely different aileron configurations, the stick forces at a given speed may vary considerably for different aileron configurations capable of providing the same value of $pb/2V$; or, conversely, for a given stick-force requirement, the speed at which that requirement can be met depends largely on the aileron configuration. This is illustrated in figure 6. Figure 6 is presented in a form similar to that of the preceding figure, and the solid curves which represent total aileron deflection of 20° and 40° were taken from the preceding figure. The dashed curves show the speeds at which a stick force of 10 pounds is required to give a $pb/2V$ of 0.07 for a typical airplane with no aerodynamic balance on the ailerons. In order to meet this requirement at a speed higher than that given by the dashed curves some aerodynamic balance must be used. The wing span of the airplane is 35 feet and the aspect ratio is 8. The stick-force value of 10 pounds is rather arbitrary. A force as high as 30 pounds may be acceptable, but a much lower force is desirable provided the friction is not so great as to prevent the ailerons from returning to their trim position if they are deflected and then released.

It should be noted that the speeds at which the requirement of a force of 10 pounds for a $pb/2V$ of 0.07 can be met are considerably lower for the wide-chord, short-span ailerons than for the narrow-chord, long-span ailerons. For example, if there is no objection to the use of an aileron span of 70 percent of the wing semispan, the requirement can be met at 150 miles per hour with ailerons having a chord of 14 percent of the wing chord and a total deflection range of 20°. For most light airplanes, there would be no need for aerodynamic balance with such an aileron configuration. If a lift flap is to be employed on the airplane, the aileron span probably would have to be shortened considerably. Assume, for example, that the aileron span is to be shortened to

35 percent of the wing semispan. This will necessitate an increase in the deflection range, the aileron chord, or possibly, increases in both. If the aileron chord is maintained constant, the deflection range would have to be increased to about 40°. For this configuration, the requirement of a force of 10 pounds for a $pb/2V$ of 0.07 could be met up to a speed of about 100 miles per hour. Aerodynamic balance would be needed if the requirement is to be met at a higher speed. If the deflection range is maintained at 20°, the aileron chord would have to be increased to about 37 percent of the wing chord in order to obtain the desired aileron span. For this configuration, a stick force of 10 pounds for a $pb/2V$ of 0.07 would occur at about 65 miles per hour. It is evident from the foregoing example that aerodynamic balance may be needed if short-span ailerons are used.

Lateral-control devices that can be used in conjunction with full-span lift flaps have been of interest for many years. Spoiler-type devices seem best suited to such arrangements. Figure 7 shows four such devices. Data on these and other devices suitable for use with full-span flaps are contained in reports that now are being released. (See references 9 to 18.) A collection (reference 19) has been made of the most pertinent information given in the various reports. The first two of the devices shown in figure 7 - the hinged-plate spoiler and the retractable-arc spoiler - are considered to have undesirable characteristics in that small projections of these devices above the wing surface may produce little or no rolling moment while slightly larger projections may cause very large rolling moments; and, unless the devices are located very far back on the wing, a noticeable time lag may occur between operation of the device and the response caused by that operation. In these respects the slot-lip aileron is satisfactory with flaps down. The slot-lip aileron as shown is not well adapted for use with flaps retracted, however, because the position of the retracted flap prohibits any downward movement of the aileron and therefore a rather complicated linkage would be required for its operation. In one full-scale installation (reference 11) of slot-lip ailerons on an airplane, they were used only with the flaps deflected. Conventional ailerons on the trailing edges of the flaps were used with the flaps retracted. The plug aileron (references 15 to 18), when used with a slotted flap, seems to have satisfactory characteristics with flaps either down or retracted. This device is designed in such a manner that a slot through the wing is opened as the plug is projected above the wing surface. The use of the slot seems to overcome the objectionable characteristics of the hinged plate and the retractable-arc spoilers.

In addition to the elevator, the rudder, and the aileron, a device capable of controlling the glide path may become important for light aircraft. Such a device may take the form of a simple lift flap, a spoiler, or flat plates projected into the air stream to add parasite drag. Very little work has been done so far on the development of devices specifically for glide control, but it is felt that the available information on lift flaps, spoilers, and various kinds of aerodynamic brakes (reference 20) can be utilized in the design of glide-control devices.

SYMBOLS

$\partial C_h/\partial \alpha$	variation of hinge-moment coefficient with angle of attack
$\partial C_h/\partial \delta$	variation of hinge-moment coefficient with control deflection
C_h	hinge-moment coefficient $(H/qc_s{}^2b_s)$
α	angle of attack, degrees
δ	control-surface deflection, degrees
H	hinge moment, foot-pounds
q	dynamic pressure, pounds per square foot
c_s	root-mean-square chord of control surface behind hinge line, feet
b_s	span of control surface at hinge line, feet
$pb/2V$	helix angle developed in roll, radians
p	rolling velocity, radians per second
b	wing span, feet
V	airspeed, feet per second
δ_u	upward aileron deflection, degrees
δ_d	downward aileron deflection, degrees

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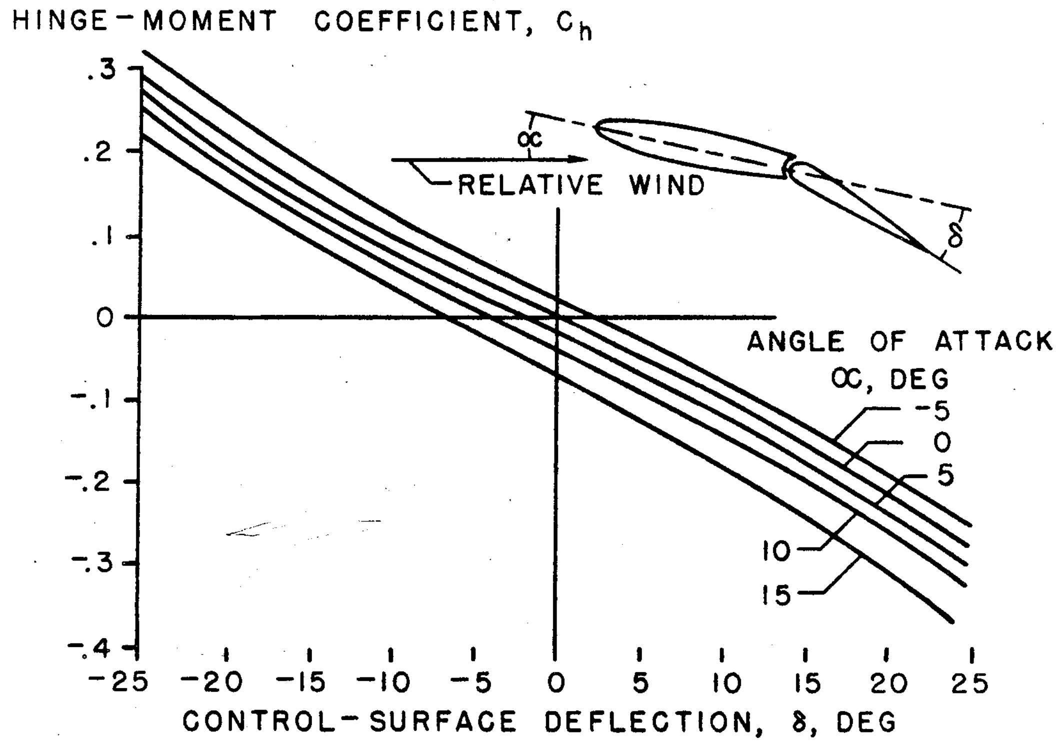

Figure 1.- Hinge-moment characteristics of a typical control surface.

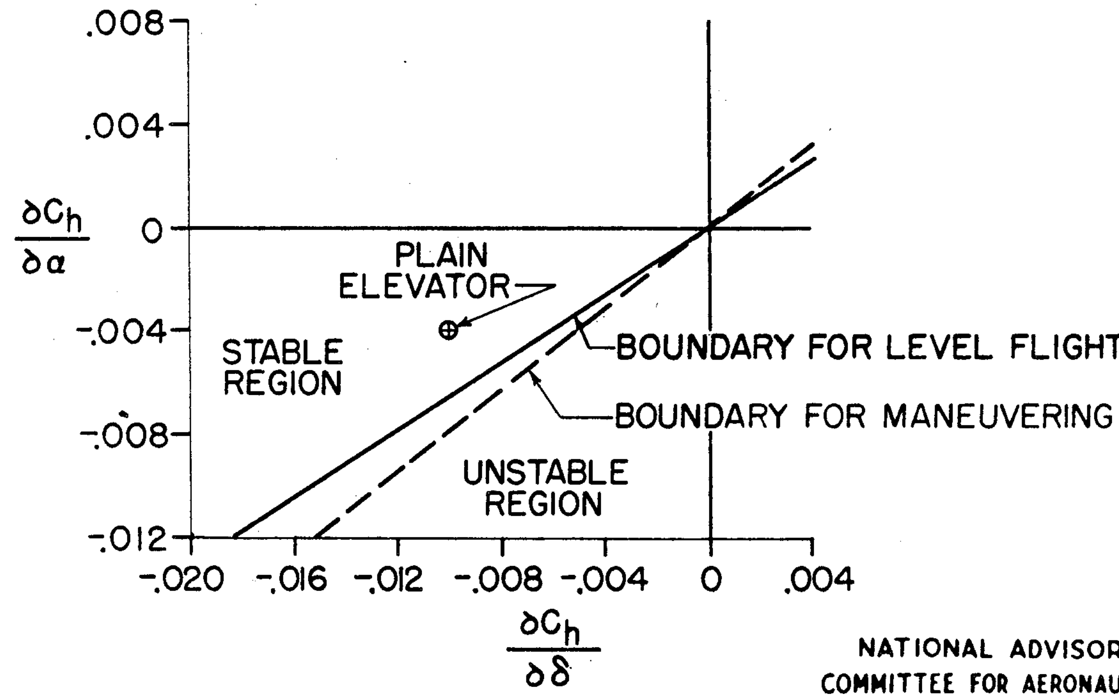

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Figure 2.- Relation of two hinge-moment parameters to stability boundaries for elevator control.

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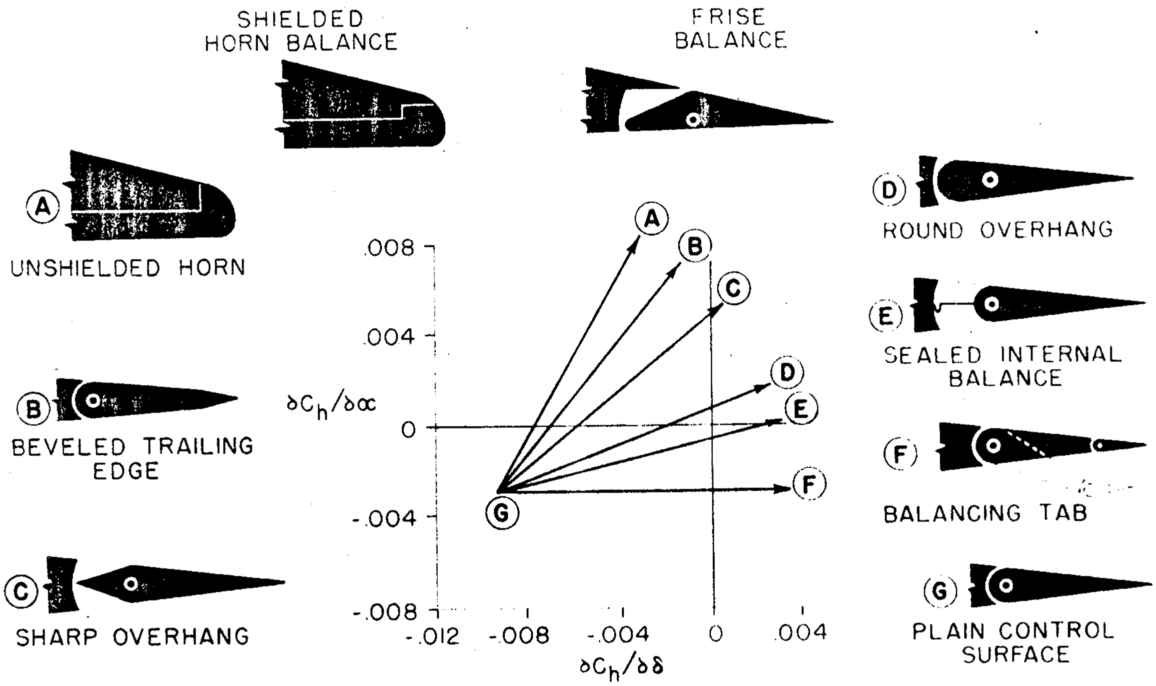

Figure 3.- Control-balance arrangements and their effects on hinge-moment parameters

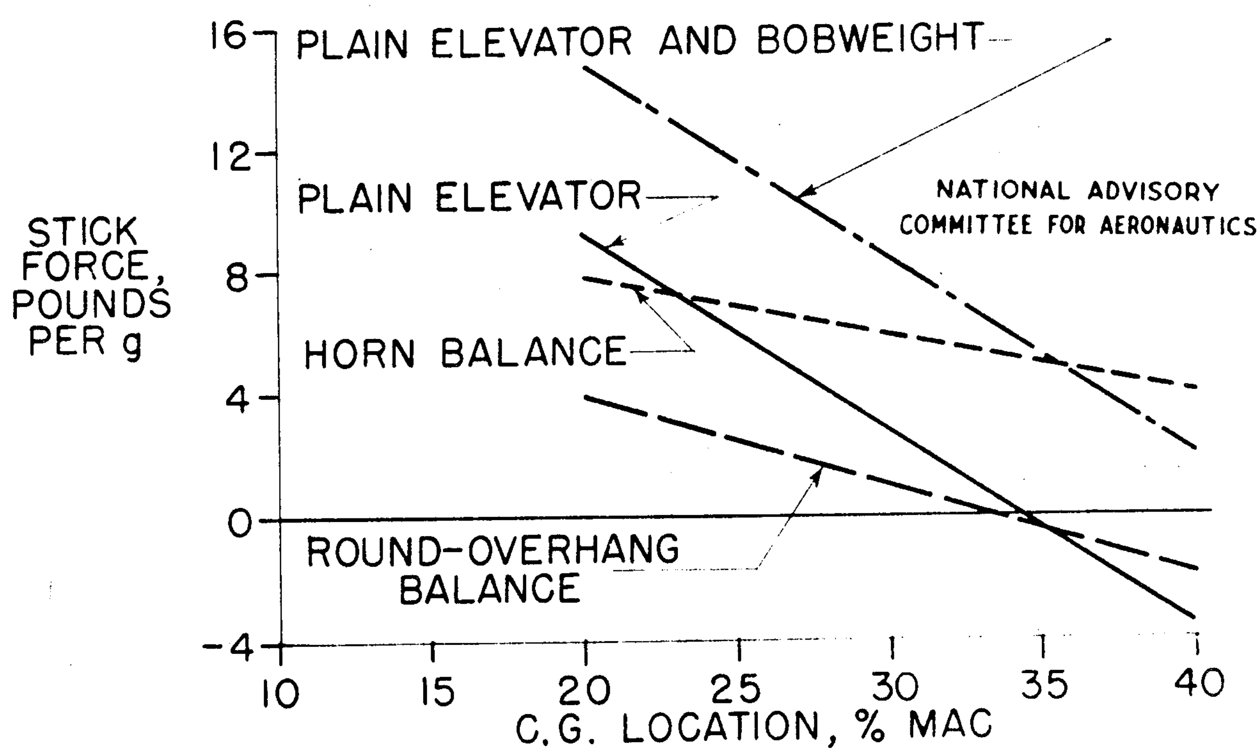

Figure 4.- Effects of elevator balance and of a bobweight on elevator stick forces in accelerated maneuvers for a typical light airplane.

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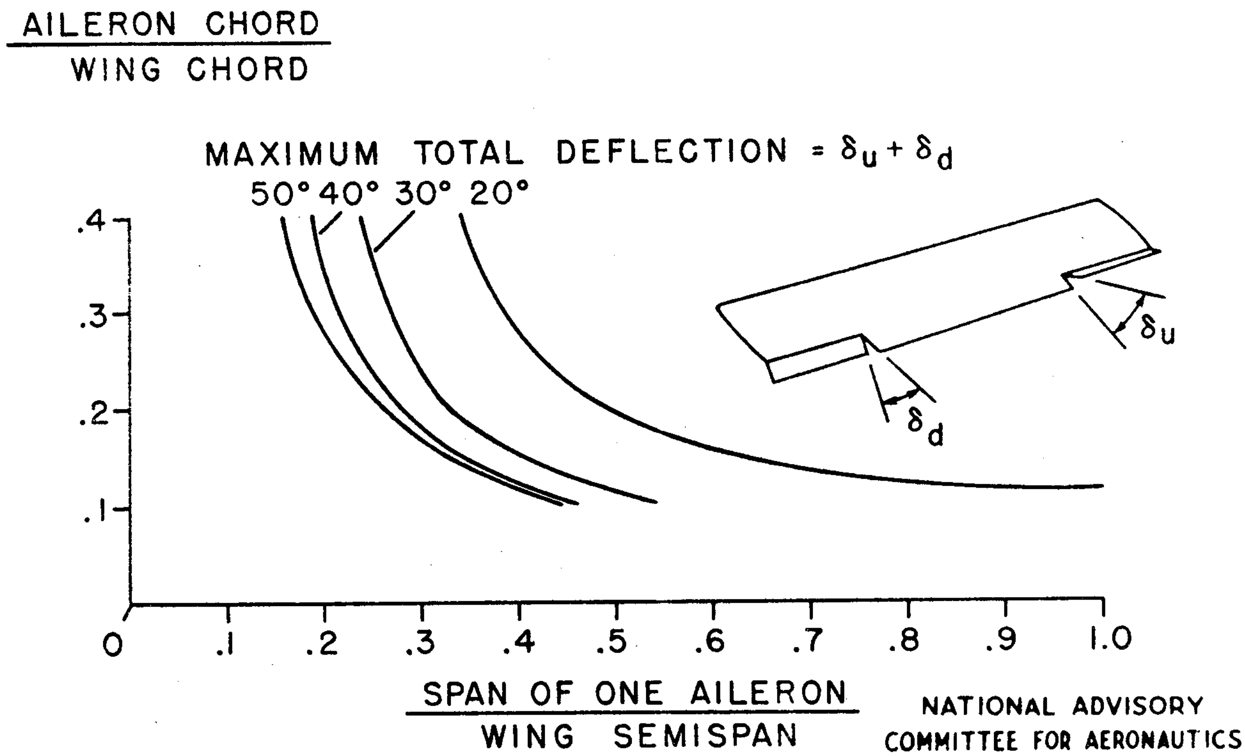

Figure 5.- Aileron configurations capable of providing a value of pb/2V equal to 0.07.

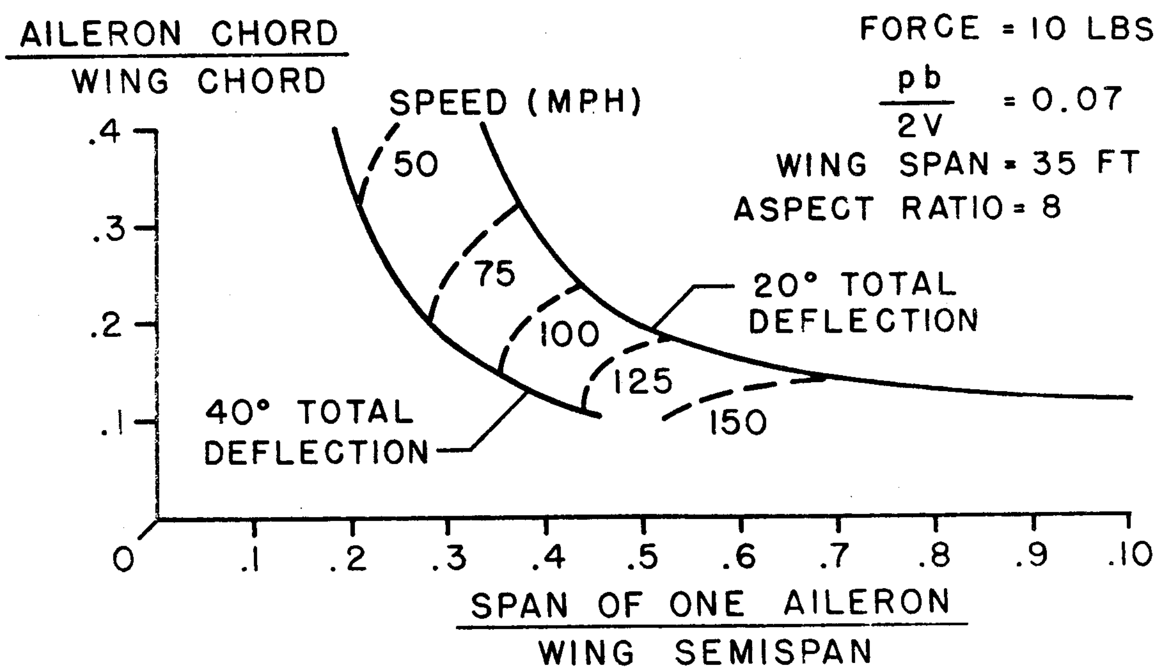

Figure 6.- Configurations of plain ailerons capable of providing at various speeds a value of pb/2V equal to 0.07 with a stick force of 10 pounds for a typical light airplane.

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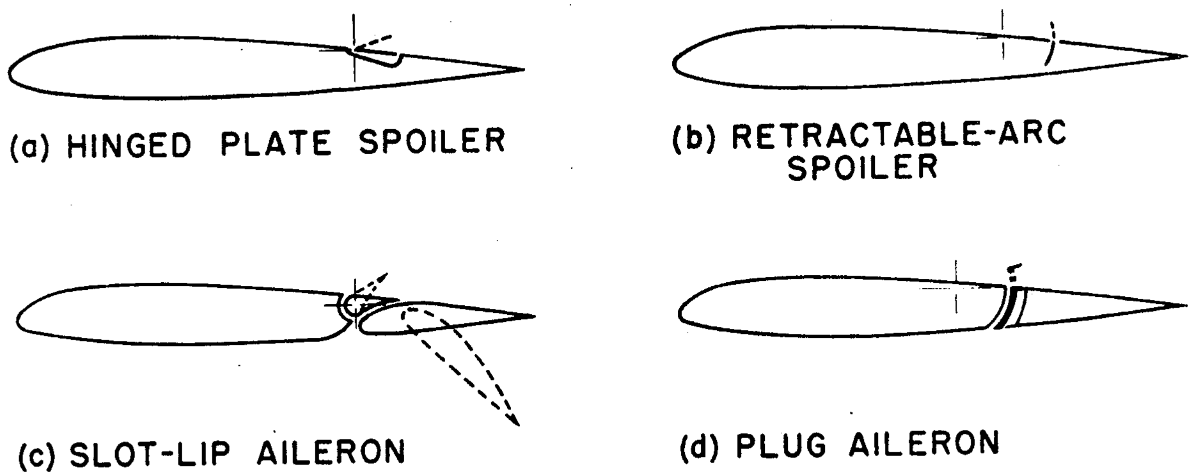

Figure 7.- Spoiler-type lateral-control devices suitable for use with full-span flaps.

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PREDICTION OF CONTROL-SURFACE

HINGE-MOMENT CHARACTERISTICS

By Charles W. Frick, Jr.

It is the purpose of this paper to present a general summary of the research the NACA is conducting on the prediction of control forces and to show how some of the results may be applied to the problem of stick-free stability for light aircraft.

The analysis that is presented is based on recent results of research on this problem being done here at the Langley Laboratory and also at the Ames Laboratory.

The problem of predicting the hinge-moment characteristics of control surfaces of finite aspect ratio from section data has been under study in the laboratories of the NACA for a number of years. It is the basic purpose of this research to develop a method of predicting control-surface hinge-moment characteristics from basic data that will account for the wide variations in aspect ratio, taper ratio, control-surface chords, and airfoil sections which aircraft designers might use.

Several years ago the NACA published a method of hinge-moment prediction which was based on what is known as "lifting-line theory." (See references 1 and 2.) In this theory the vortices which represent the lifting capacity of the tail surface are concentrated in a single line, usually the 25-percent chord line of the surface, with trailing vortices being shed as the tip is approached. This is shown in figure 1 by the vortex pattern on the left. Prediction of hinge-moment characteristics using this vortex pattern were found to be of satisfactory accuracy for preliminary design especially for horizontal tails of aspect ratio greater than 4.0. Comparisons of hinge-moment parameters predicted by this method with experimental results for aspect ratios less than 4, however, showed some error in the predictions which became worse as the aspect ratio decreased. It was found that the error was of appreciable magnitude, especially insofar as the prediction of variation of control hinge moment with tail angle of attack was concerned. Since this parameter greatly influences the loss in stability due to freeing the controls, the need for improving the accuracy of prediction was keenly felt It was surmised that the error in prediction was caused by the fact that some change in the chordwise distribution of lift, primarily a function of aspect ratio, was being neglected.

With the aid of a new application of the vortex theory of wings, called lifting-surface theory, developed by Miss Doris Cohen of the Langley Laboratory (reference 3), a further analysis of the problem was made. Essentially, this analysis uses the vortex pattern shown on the right of figure 1 wherein the vortices representing the lifting capacity of the surfaces are distributed over the tail in proportion to the local lift. With this vortex pattern, the induced downwash is calculated by the method of Miss Cohen at a number of points on the tail by summing up the contribution of all vortices. If the downwash as determined varies along the chord of the wing and the streamlines do not conform to the profile, the lift distribution along the chord of the wing is altered with a resulting change in hinge-moment-producing load over the control surface.

Figure 2 shows a comparison of the change in the chordwise loading resulting from a change in angle of attack in going from infinite aspect ratio to aspect ratio 3.0 for lifting-line and lifting-surface theory. The change is indicated by the cross-hatched area. Comparing the cross-hatched areas for lifting-line and lifting-surface theories, it can be seen that the lifting-surface theory simply provides a correction to the chordwise loading given by lifting-line theory as shown by the lower cross-hatched area of the figure on the right of figure 2. Knowing the change in chordwise loading due to change in aspect ratio, we can, of course, compute the change in hinge moment. At an aspect ratio of 5, the lifting surface-theory correction amounts to only 20 percent of the total change in the variation of control hinge moment with angle of attack while at aspect ratio of 3, it amounts to 60 percent. The expression for C_{h_α}, that is, the variation of control hinge moment with tail angle of attack from lifting-surface theory is given by the equation shown below the diagram on the right of figure 2. Comparisons between the values of C_{h_α} determined by this equation and experimental data show remarkable agreement.

The change with aspect ratio in the chordwise loading due to control deflection is shown on figure 3. When the control is deflected two types of chordwise loading are added, the basic control lift associated with the change in airfoil camber due to deflection of the control which has a peak over the hinge of the control, and the airfoil additional lift resulting from an effective change in the angle of attack. (See reference 4.) Analysis of the vortex pattern for this type of chordwise loading has shown that the lifting-surface correction to the chordwise loading is nearly the same as that which would result if the increase in lift due to control deflection were obtained by simply increasing the tail angle of attack. In other words, the change in chordwise loading due to aspect-ratio change is of the same form for lift produced by control deflection as it is for lift obtained by change in angle of attack. Fortunately, this means that the equation shown on figure 3 expressing the relationship between hinge-moment characteristics for aspect ratio ∞, that is, profile characteristics, and for any other aspect ratio has the same form for both lifting line and lifting-surface theory except for an additional term $(\Delta C_{h_\delta})_{LS}$, which is usually of very small magnitude.

From this equation it may be seen that the aspect-ratio correction to C_{h_δ} is always less than the aspect-ratio correction to C_{h_α} so that as the tail aspect ratio decreases we may expect C_{h_α} to increase, that is, become less negative, more rapidly than C_{h_δ}. This is shown by figure 4. The hinge-moment parameters shown are those for a 30-percent chord elevator on a tail plane of NACA 0009 airfoil section and elliptical planform. As the aspect ratio of the tail decreases, C_{h_α} approaches a value of zero at aspect ratio 2.5. C_{h_δ} is affected to a smaller extent. You will note that the difference between lifting-line and lifting-surface theories become of significance at low aspect ratios.

This new method of control-surface hinge-moment prediction is of very great use to designers. Figure 5 shows an example of how it may be applied to the estimation of stick-free stability. The results shown have been calculated for constant stick-fixed stability. The curve for the unbalanced elevator may be considered to be that for a personal aircraft of, say, 2500 pounds gross weight. As you can see, changing the aspect ratio of the horizontal tail from 5.0 to 3.0 results in an increase in stick-free stability of 0.03, that is, a rearward shift of the neutral point of 3 percent of the mean aerodynamic chord. This amounts to 20 percent of the stick-fixed stability of the assumed airplane.

For a larger aircraft of, say, 5000 pounds, where control balance is required to reduce maneuvering and landing stick forces, the effect is more powerful. This is shown by the curve labeled 35-percent-chord elevator balance. For this airplane, changing the aspect ratio of the tail from 5.0 to 3.0 results in a gain in stick-free stability of 0.075 or a rearward shift of the neutral point of 7.5 percent of the mean aerodynamic chord. This amounts to one-half of the stick-fixed stability of this airplane.

It is evident, therefore, that this method of control-surface hinge-moment prediction gives the designer a means of determining the basic geometric characteristics of the horizontal tail angle of attack. In other words, the change in chordwise loading due to aspect-ratio change is of the same form for lift produced by control deflection as it is for lift obtained by change in angle of attack. Fortunately, this means that the equation shown on figure 3 expressing the relationship between hinge-moment characteristics for aspect ratio ∞, that is, profile characteristics, and for any other aspect ratio has the same form for both lifting line and lifting-surface theory except for an additional term $(\Delta C_{h_\delta})_{LS}$, which is usually of very small magnitude.

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It is evident, therefore, that this method of control-surface hinge-moment prediction gives the designer a means of determining the basic geometric characteristics of the horizontal tail necessary for the attainment of desirable control-free stability and desirable control force gradients.

SYMBOLS

C_{h_α} variation of control hinge-moment coefficient with tail angle of attack

C_{h_δ} variation of control hinge-moment coefficient with control deflection

C_m pitching-moment coefficient (M/qcS)

C_L lift coefficient (L/qS)

$\frac{P}{C_L}$ normal pressure per unit lift coefficient

AR aspect ratio

M pitching moment, foot-pounds

q dynamic pressure, pounds per square foot $\left(\frac{1}{2}\rho V^2\right)$

c mean aerodynamic chord of wing, feet

S wing area, square feet

L lift, pounds

V airspeed, feet per second

ρ density of air, slugs per cubic foot

α angle of attack, degrees

K constant

Subscript:

LS lifting surface

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2. Crane, Robert M.: Computation of Hinge-Moment Characteristics of Horizontal Tails from Section Data. NACA CB No. 5B05, 1945.

3. Cohen, Doris: A Method for Determining the Camber and Twist of a Surface to Support a Given Distribution of Lift. NACA TN No. 855, 1942.

4 Allen, H. Julian: Calculation of the Chordwise Load Distribution over Airfoil Section with Plain, Split, or Serially Hinged Trailing-Edge Flaps NACA Rep. No. 634, 1938.

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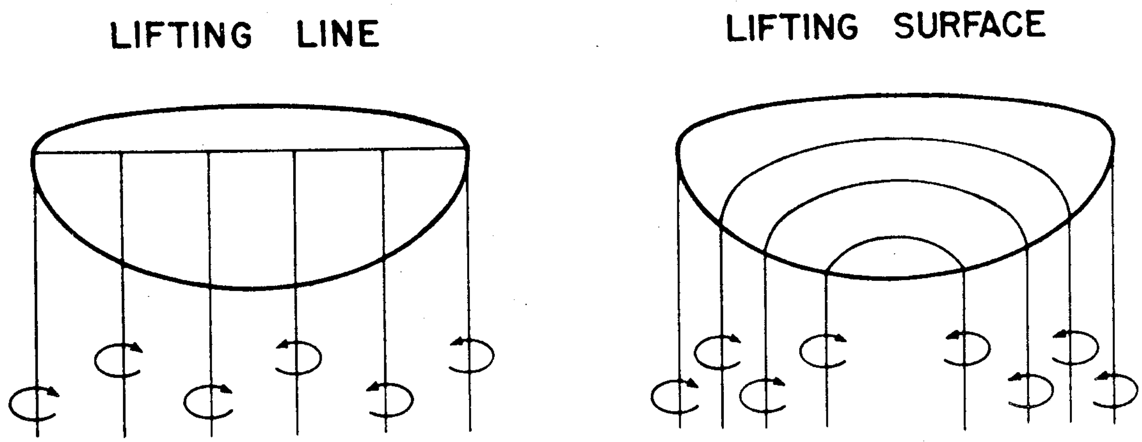

Figure 1.- Horizontal-tail vortex patterns.

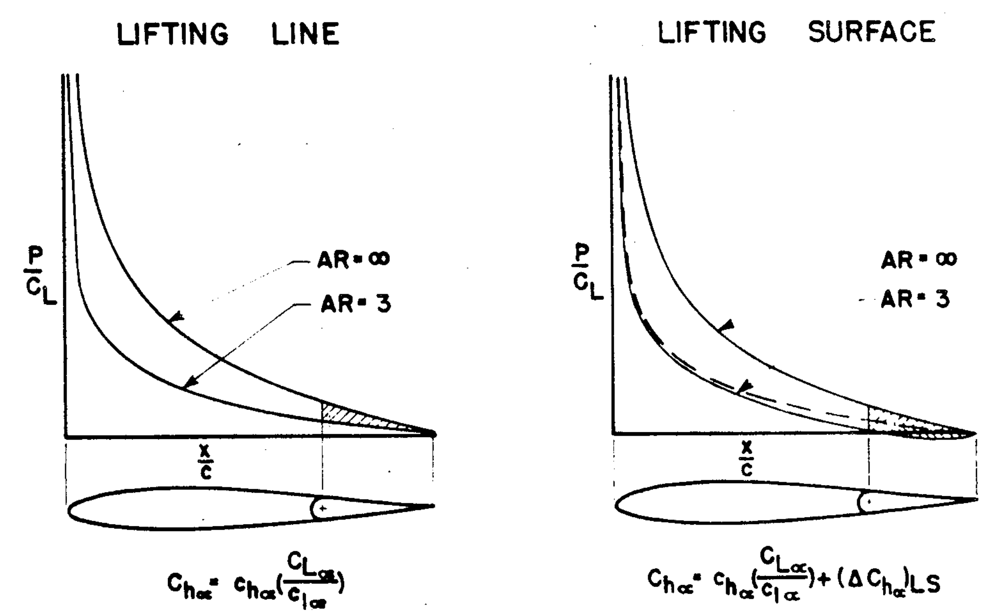

Figure 2.- Correction to the variation of elevator hinge-moment with tail angle of attack due to change in aspect ratio.

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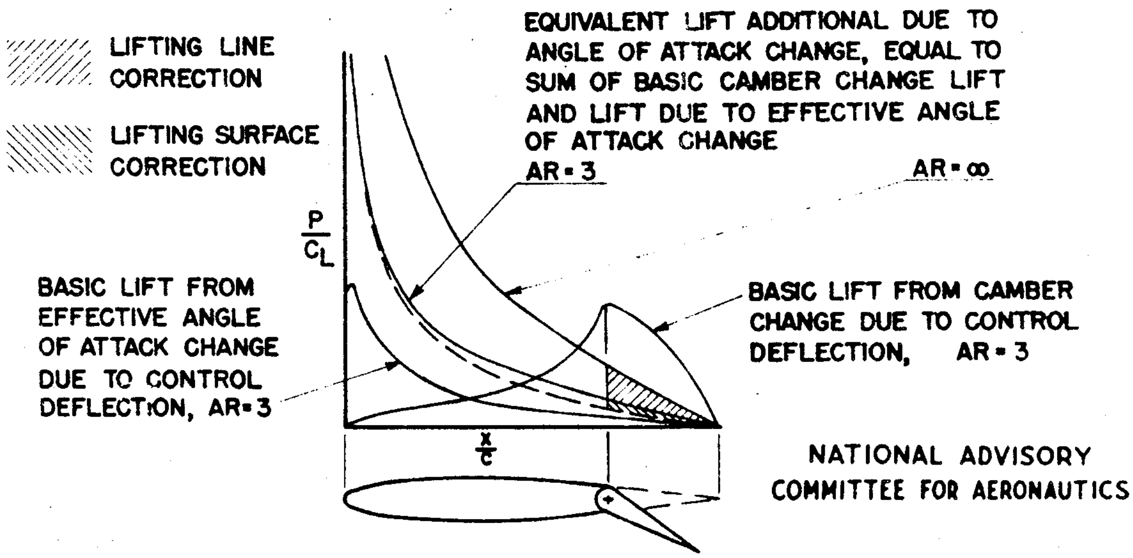

$$C_{h_\delta} = c_{h_\delta} + \alpha_\delta\,(c_{h_\alpha} - C_{h_\alpha}) + (\Delta C_{h_\delta})_{LS}$$

Figure 3.- Correction to the variation of elevator hinge-moment with elevator deflection due to change in aspect ratio.

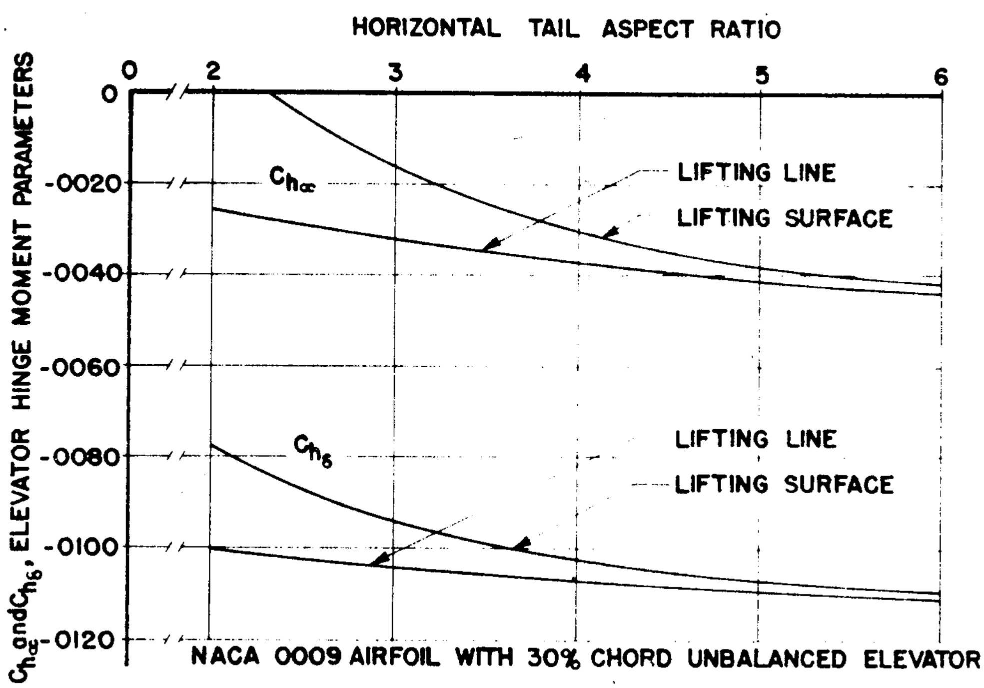

Figure 4.- Variation of elevator hinge-moment parameters with horizontal-tail aspect ratio.

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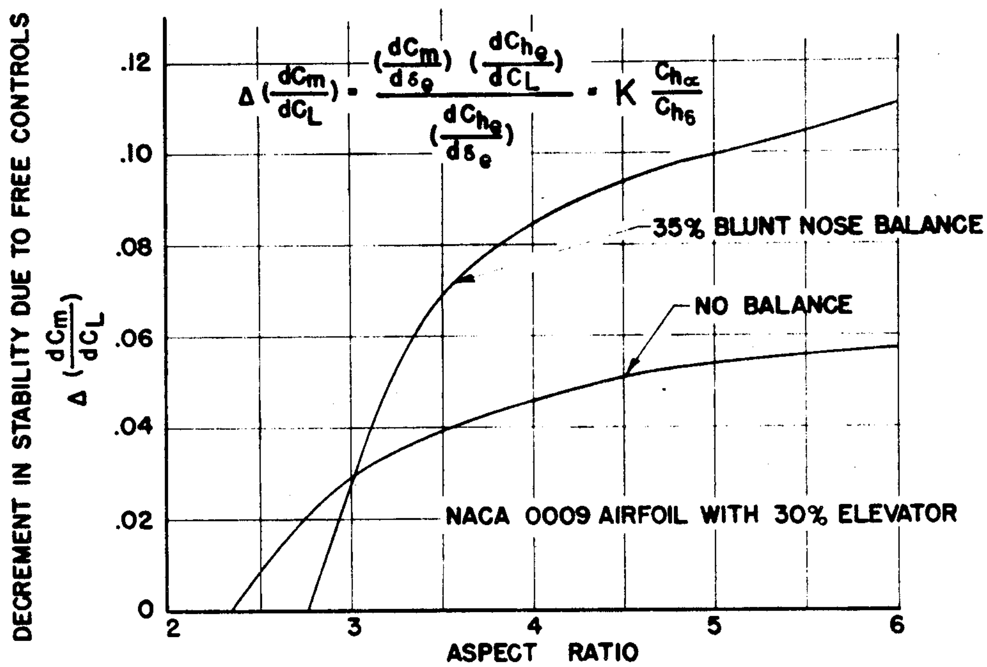

Figure 5.- Variation with aspect ratio of the decrement in stability due to free controls.

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SAFETY AND SPINNING

A FLIGHT INVESTIGATION TO INCREASE THE SAFETY OF A LIGHT AIRPLANE

By Paul A. Hunter

During 1940 and 1941 a series of modifications were incorporated in a light airplane to increase its safety. It was intended to hold the changes to a minimum, to make them simple enough to be readily incorporated in airplanes already in service, and not basically to alter the appearance of the airplane.

The method originally chosen to make the airplane stallproof was to eliminate the effect of power on the elevator angle required to stall and limit the up-elevator travel to some value below that required to stall. While this was not completely achieved, a definite step in this direction was taken.

The final result of this investigation was an airplane which was spinproof, could not be stalled in turns, and while not stallproof in straight flight, possessed superior stalling characteristics as compared with the original airplane. Figure 1 shows the airplane as it appeared originally together with the modifications which were made to it. A few of these changes are immediately apparent but all the changes will be described in detail later.

Figure 2 shows the stick-fixed static longitudinal stability of the original and modified airplanes. The increased stability of the modified airplane is apparent despite the adverse effect of its more rearward center-of-gravity position. The reduction of the effect of power is also apparent. The stick force characteristics of the modified airplane likewise showed a reduction in the effect of power on the stick forces.

Stalls from straight and level flight with the original airplane resulted in rapidly diverging lateral oscillations which could not be controlled by the ailerons. The stalling characteristics of the modified airplane were considered superior to those of the original airplane in that the stall, both with power on and power off, consisted merely of a mild dropping of the nose. The region in which the lateral instability occurred for the original airplane and in which the longitudinal instability occurred for the modified airplane was defined by the up-elevator position. Elevator angles above which instability occurred are shown by the curves of figure 3. Note the difference between the power-on and power-off conditions, 12° for the original airplane and 3° for the modified airplane. As was mentioned before, the effect of power was not completely eliminated as planned but a large reduction was effected. The center-of-gravity position for a given loading of the airplane was moved back considerably because of the increased weight of the tail but the range of loadings was not affected.

In turning flight the instability associated with the complete stall in the original airplane was essentially the same as from straight flight. The violence of all motions accompanying the stall was increased in turning flight somewhat because of the effectively increased wing loading under accelerated conditions. The original airplane could be stalled out of a turn with power on or off. It was possible with the modified airplane to pull the stick all the way back in turning flight without developing any uncontrolled-for motions either with power on or power off. The stall could not be reached because of the additional up-elevator deflection required to produce pitching velocity in the turn.

It was not practicable to limit the up-elevator deflection to the extent required to make the modified airplane stall-proof in straight flight because of the up-elevator deflection required to make a 3-point landing. The average up-elevator deflection required for a 3-point landing was 27.5° for a center of gravity of 30 percent of the mean aerodynamic chord and was reduced 10° as the center of gravity moved back 7 percent.

Returning to figure 1 it can be seen that 7 final modifications were incorporated in the airplane in producing these characteristics. The incidence of the wing was changed from 2° to -1.2° and the washout was increased from 3° to 5°. The thrust axis was depressed 7° and the rudder travel was limited to ±15°. The elevators were moved out of the propeller slipstream and the areas and aspect ratios of the horizontal and vertical tails were increased. Depressing the thrust axis, reducing the incidence, and moving the elevators out of the propeller slipstream all tended to increase the elevator angle required to stall with power on. Increasing the wing washout eliminated the lateral instability at the stall. Increasing the aspect ratio of the horizontal and vertical tails improved the effectiveness of the surfaces while the added area increased the static longitudinal stability and the directional stability of the airplane. It was found that the airplane incorporating all the changes except limiting the rudder travel to ±15° could be held in a steady spin to the left. If the left rudder was decreased to 15° after a steady state of rotation was obtained, the airplane would recover from a spin after several turns with either power on or power off. The rudder travel was therefore limited to ± 15° and all attempts to produce a spin failed. The modified airplane with the limited rudder could be taxied satisfactorily with the use of brakes and no loss of directional control during the take-off was encountered. The maximum angle of bank that could be obtained in a steady sideslip was reduced to about half that of the original airplane because of the limited rudder travel, but the rudder was still sufficient to counteract the adverse yaw of the ailerons.

The characteristics of this modified airplane may be summarized as follows:

Motions accompanying the stall were greatly reduced. The resulting longitudinal motion was considered less dangerous than the lateral instability encountered on the original airplane. In turns the stick was held full back without producing a stall.

The airplane could not be spun, power on or power off, with any setting of the controls. The general flying characteristics were not materially altered in normal flight.

Hunter, Paul

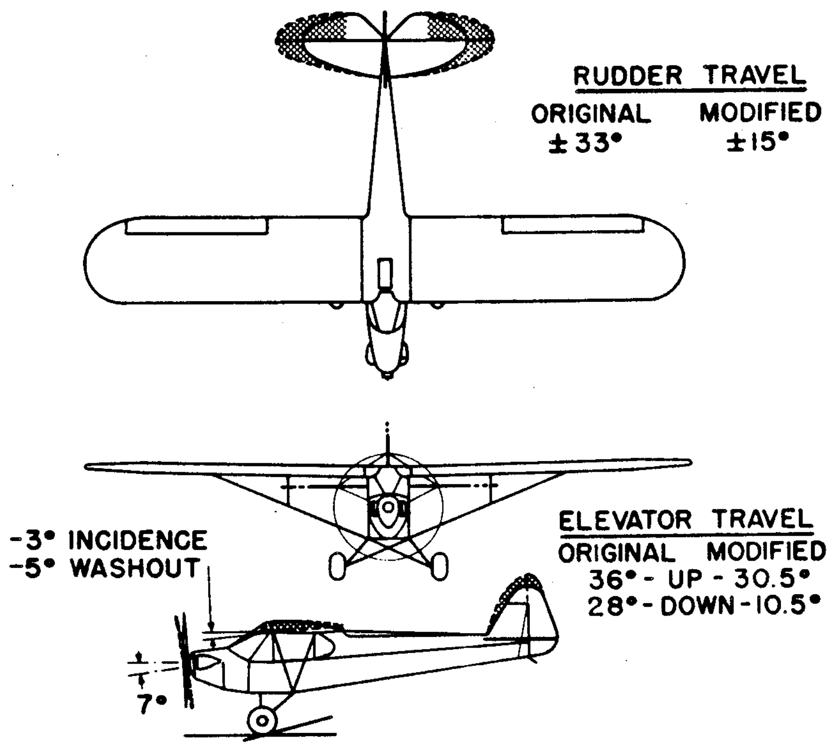

Figure 1.- Three-view drawing showing modifications to original airplane.

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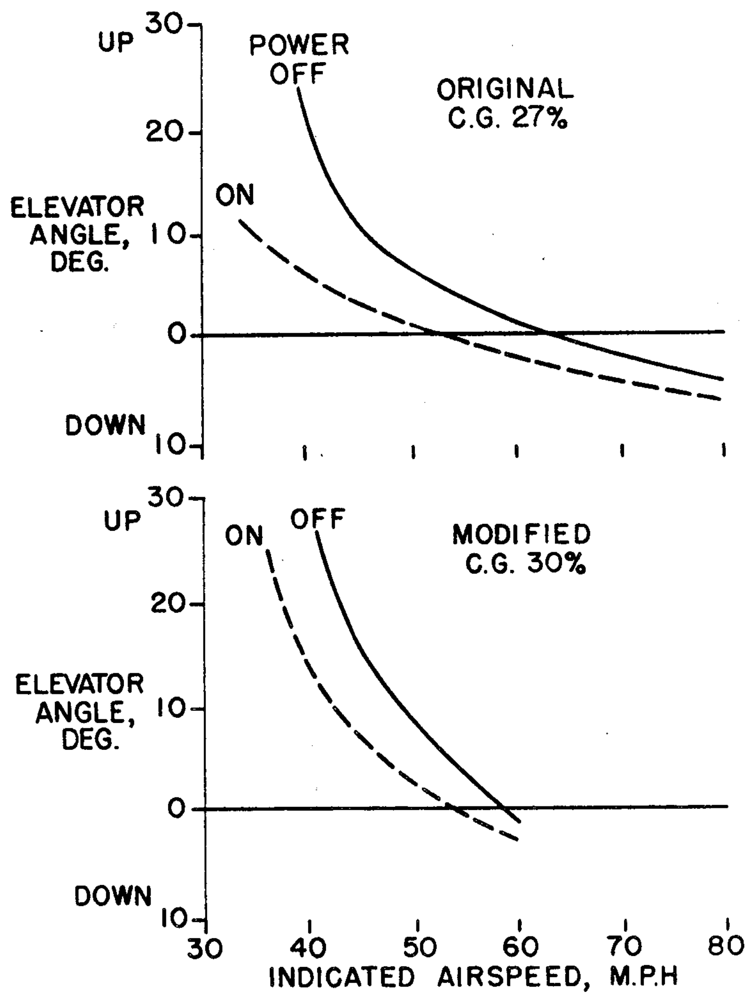

Figure 2.- Stick-fixed static longitudinal stability characteristics of the original and modified airplanes.

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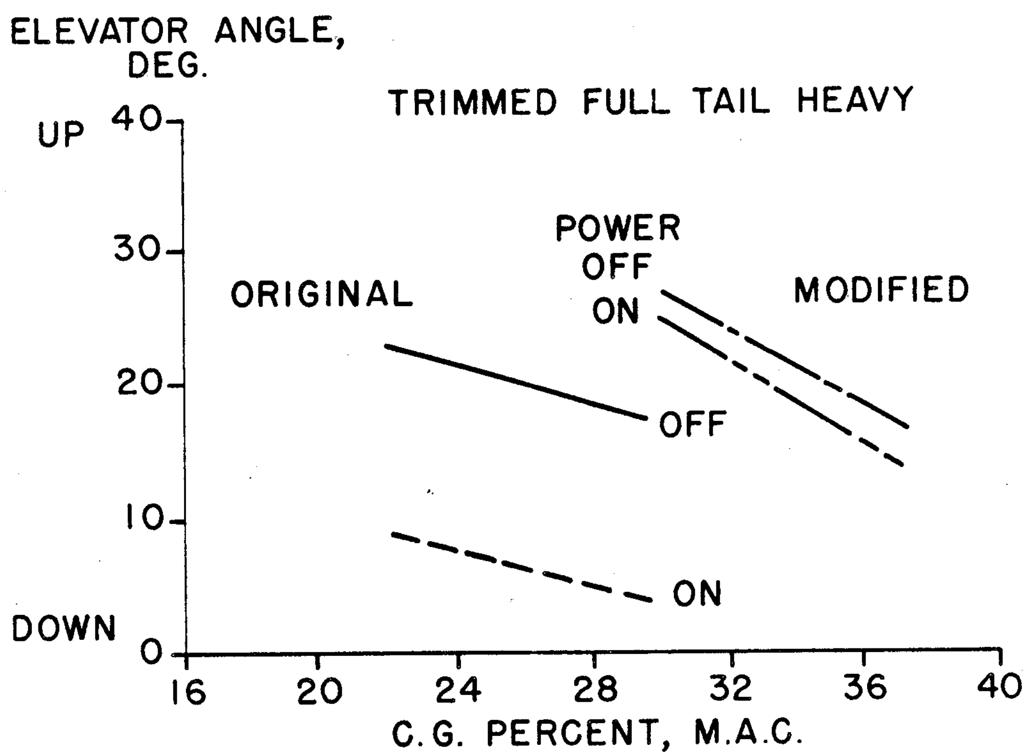

Figure 3.- Elevator angles above which instability existed for the original and modified airplanes.

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FACTORS AFFECTING SPINNING OF LIGHT AIRPLANES

By Anshal I. Neihouse

The spin problem for the personal-owner or light airplane may be considered as twofold: (a) elimination of the incipient spin, and (b) elimination of, or satisfactory recovery from, the fully-developed spin.

Mr. Hunter has just discussed elimination of the stall in turning flight, which appears desirable for preventing incipient spins at low altitudes. His discussion also included elimination of the spin of a light airplane by limiting the rudder travel. In this connection, tests of numerous models in the NACA spin tunnel have indicated that the rudder is very often the predominant control maintaining the spin.

During the war years, spin investigations have been conducted in the NACA spin tunnel on approximately 150 different military airplane designs to determine recovery characteristics from developed spins. From the results of the investigations, it has been shown, and reported in reference 1, that an airplane in a fully developed spin will recover rapidly if sufficient and effective control has been provided in its design. Quite a few of the designs tested had proportions of mass and dimensional characteristics that simulated those of airplanes in the personal-owner category and accordingly the results from tests of military airplanes have been applied to light airplanes. In order to provide effective control for satisfactory recovery from spins of personal-owner airplanes, a criterion has been set up which is an extension of the one previously reported in reference 1 and is shown in figure 1. This criterion combines the same factors previously found important in effecting recovery from the spin. As before, for different values of relative density, values of tail-damping power factor have been plotted against a nondimensional expression for the difference in moments of inertia about the airplane X- and Y-axes, and regions of satisfactory and unsatisfactory recoveries have been defined. These regions are based on results of spin investigations of many models in the NACA spin tunnel. Before discussing this figure more thoroughly, I should first like to tell you a little about the factors that have been plotted.

Tail-damping power factor (TDPF) is an indication of the effectiveness of the vertical tail in a spin. The method of computing it has been fully explained in reference 1 and is illustrated in figure 2. By assuming a spin at an average angle of attack of 45°, the flow at the tail beyond the horizontal surfaces is assumed to diverge 15° at each end. The wake thus defined indicates the rudder area which is shielded by the horizontal tail. The tail-damping power factor is made up of the two terms indicated, the first being a damping term and the second being a rudder-effectiveness term. The damping term depends on the fixed area below the horizontal tail and its distance from the center of gravity of the airplane. The rudder-effectiveness term depends on the unshielded rudder area and its distance from the center of gravity.

The airplane relative density factor $\mu = \frac{W}{\rho g S b}$ is the ratio between the so-called density of the airplane (W/gSb) and the density of the surrounding air (ρ) in which the airplane is moving. It may be said to be an indication of the interaction of the inertia and aerodynamic forces and moments acting. In order to determine the values of relative density (μ) typical of light airplanes, a range of wing loadings (W/S) has been plotted against a range of spans (b) considered representative and the corresponding values of μ have been indicated in figure 3 for sea-level density. For wing spans of 29 to 40 feet and wing loadings from approximately 6 to 24 pounds per square foot, the corresponding values of μ for sea-level air density varied from approximately 2 to 8.

Returning to figure 1, the term plotted as the abscissa $\frac{I_X - I_Y}{mb^2}$, the difference in moments of inertia about the X- and Y-axis, divided by the airplane mass and span squared is an indication of how the mass is distributed in the airplane, that is, whether there is relatively more mass distributed along the wings or along the fuselage. Increasing values of the parameter in a positive direction indicate increased distribution of mass along the wings.

The uppermost curve shown in figure 1 was previously presented (reference 1) for all airplanes having values of relative density up to 15. For the present investigation, curves were obtained for airplanes having values of μ of 2 to 5 and of 5 to 9. Thus from the data plotted, minimum values of vertical-tail design required insure satisfactory recovery from developed spins may be obtained for values of μ in the personal-owner airplane range.

Airplanes having values of the plotted factors falling above their corresponding curve will recover satisfactorily whereas those having values falling below the curve will not recover satisfactorily.

For the relatively very light airplanes $(\mu = 2 \text{ to } 5)$, it appears that very little vertical fin and rudder area is required to insure satisfactory recovery, especially if rudder reversal is followed by moving the stick forward. As the relative density of the airplanes increases, however, it becomes increasingly important to provide a sufficient TDPF to insure satisfactory recovery. Here too, movement of the elevator down may be of appreciable assistance in terminating the spin, but as a factor of safety, it is felt that sufficient vertical tail area should be provided to terminate the spin without assistance of the elevator. It can be noted that as the distribution of mass along the wings is increased, larger values of TDPF are required to insure recovery by rudder reversal.

Based on the requirements for vertical tail design as indicated in figure 1, it appears that proper consideration of the factors involved will lead to an airplane which will recover rapidly from the spin.

SYMBOLS

μ	airplane relative density factor $\left(\frac{13.1(W/S)}{b}\right.$ at sea level$\left.\right)$
W	weight of airplane, pounds
S	wing area, square feet
b	wing span, feet
ρ	density of air, slugs per cubic foot
I_X	moment of inertia of airplane about X body axis (fuselage axis)
I_Y	moment of inertia of airplane about Y body axis (wing axis)
m	mass of airplane $\frac{W}{g}$
g	acceleration of gravity, feet per second^2
TDPF	tail-damping power factor
L	distance between center of gravity of airplane and centroid of fixed area F
L_1	distance between center of gravity of airplane and centroid of area R_1

L_2 distance between center of gravity and centroid of area R_2

R_1 unshielded rudder area above horizontal-tail wake

R_2 unshielded rudder area below horizontal-tail wake

F fixed area below horizontal tail

REFERENCE

1. Neihouse, Anshal I., Lichtenstein, Jacob H., and Pepoon, Philip W.: Tail-Design Requirements for Satisfactory Spin Recovery. NACA TN No. 1045, 1946.

Neihouse

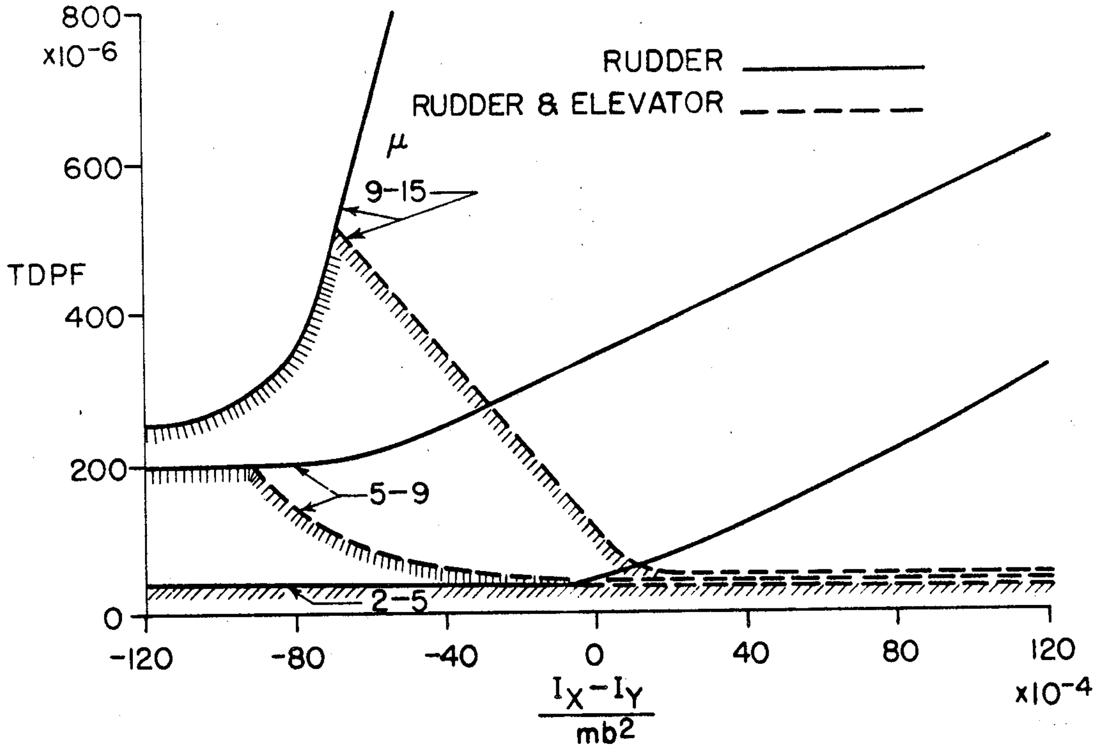
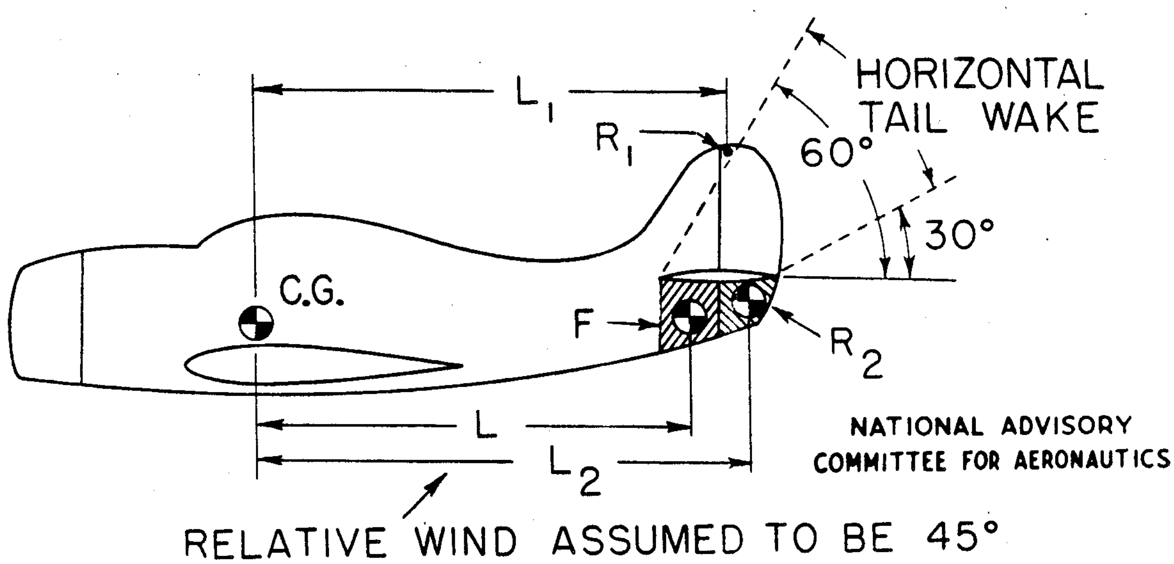


Figure 1.- Vertical-tail design requirements for personal-owner type airplanes.



$$\text{TAIL-DAMPING POWER FACTOR} = \frac{FL^2}{S(b/2)^2} \times \frac{R_1 L_1 + R_2 L_2}{S(b/2)}$$

Figure 2.- Method of computing tail-damping power factor.

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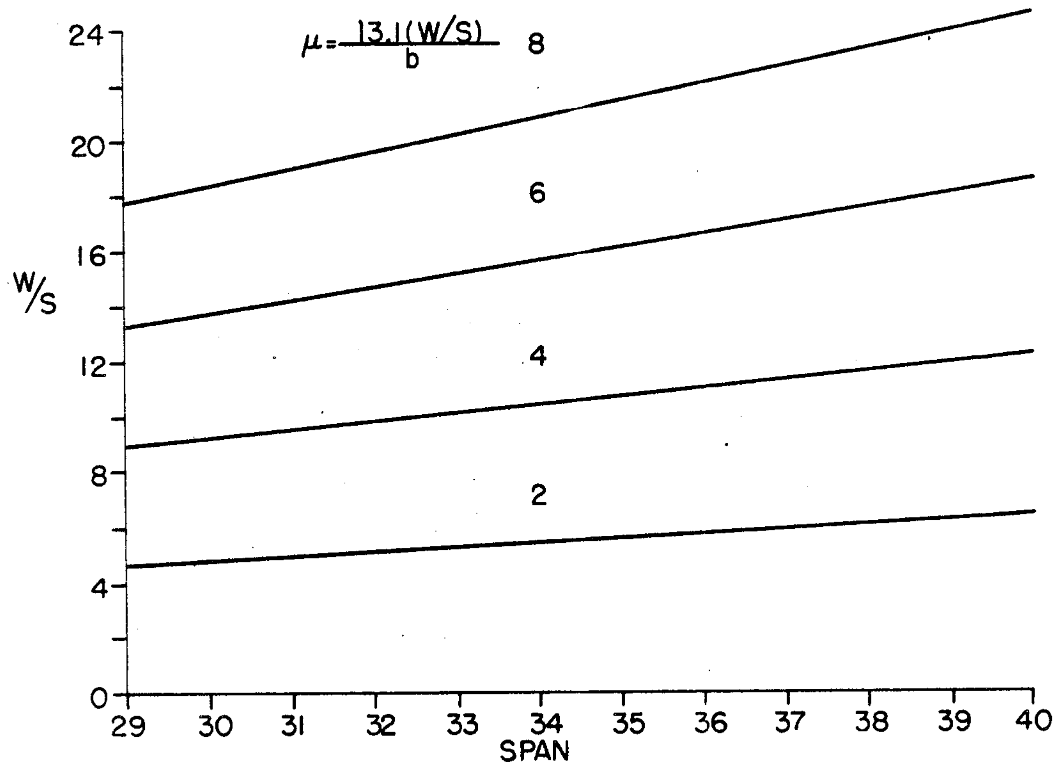

Figure 3.- Variation of relative-density factor with wing loading and span for personal-owner type airplanes.

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WING DESIGN

DEVELOPMENT OF AIRFOILS AND HIGH-LIFT DEVICES

By Laurence K. Loftin, Jr.

During the course of the war years, a new type of airfoil section, known as the NACA low-drag or 6-series section, was developed here at the Langley Laboratory. Low-drag airfoils differ from the older NACA 4- and 5-digit series airfoils, such as the NACA 2412 and 23012, in that they were theoretically derived by potential-flow methods to have pressure distributions of a type permitting extensive laminar flow in the boundary layer and thus very low profile-drag coefficients. A comparison of an NACA 23012 with two of the newer low-drag airfoil sections is shown in figure 1. Two low-drag sections, the NACA 63_1-412 and the NACA 66_1-212, are designed to permit laminar flow over the airfoil surfaces to 30- and 60-percent chord, respectively.

In order to provide the airplane designer with systematic aerodynamic data from which to choose low-drag airfoils suitable for different applications, a series of approximately 100 related low-drag airfoils was derived and tested in a specialized two-dimensional wind tunnel permitting the attainment of full-scale Reynolds numbers and having a turbulence level approaching that of free air. The lift, drag, and pitching-moment characteristics of all the airfoils were obtained at Reynolds numbers of 3, 6, and 9 million, and, in addition, tests were made with each airfoil to determine the effect upon the aerodynamic characteristics of surface roughness.

An idea of the drag characteristics of these newer airfoil sections may be obtained from figure 2 which shows drag results for a typical low-drag airfoil section (63_1-412) of 12-percent thickness and 0.4 design lift coefficient. A design lift coefficient of 0.4 corresponds to 2-percent camber on these airfoils. These data are for a Reynolds number of 6 million which corresponds roughly to the cruising Reynolds number of a personal-owner-type airplane. The minimum drag coefficient of the low-drag section is approximately 25 percent lower than that of the older NACA 23012 shown here. Varying the amount of camber has no effect upon the value of the minimum drag but shifts the range of lift coefficients over which low drag is obtained; thus it is possible that by a proper choice of camber the minimum drag may be obtained at nearly any desired cruising lift coefficient. The width of the low-drag range increases with increasing airfoil thickness ratio with only a very slight increase in the value of the minimum drag coefficient.

These drag data are for airfoil models having perfectly smooth and fair surfaces and represent the optimum or ideal drag characteristics that may be obtained. Many airplanes operate with wings which are too dirty or gritty to permit the attainment of extensive laminar flows and thus the indicated low drag coefficients. The exact size of the permissible roughness depends upon the type of roughness, the size of the wing and the Reynolds number, and is extremely difficult to predict. In an effort to obtain data indicative of the lift and drag characteristics which might be expected for the worst conditions of surface roughness corresponding to sand or mud at the leading edge, each of the 100 airfoils was tested with its leading edge sufficiently rough to cause a fully developed turbulent layer over the airfoil surfaces.

The upper curves shown for the two airfoils with rough leading edges indicate large increases in the value of the minimum drag with little difference between the drag of the low-drag airfoil and the older NACA 23012. An analysis of all available drag data has shown that with leading-edge roughness sufficient to cause fully developed turbulent boundary layers, the minimum drag coefficient is relatively insensitive to airfoil shape and increases with airfoil thickness. Surface unfairness or waviness resulting from manufacturing inaccuracies may also prevent the attainment of the expected low drag coefficients; however, some surface irregularities may be tolerated in the range of Reynolds numbers extending up to 6 million or 8 million. A detailed discussion of this subject is contained in NACA TN-1151 reference 1.

For an airplane having 150 square feet of wing area, the 25-percent reduction in minimum profile drag coefficient resulting from the use of a low-drag airfoil instead of a conventional one amounts to a saving of 12 horsepower at 200 miles per hour. The difference in the drag of the smooth and rough low-drag section amounts to about 36 horsepower. Whether or not these savings are significant will, of course, depend upon the percentage that the wing drag is of the total airplane drag.

The variation of the maximum lift coefficient with airfoil thickness ratio and camber is shown in figure 3 for a number of NACA low-drag airfoil sections. These data are for a Reynolds number of 3 million which corresponds to the landing Reynolds number of a personal-owner-type airplane. The highest maximum lift coefficient corresponds to the 12-percent-thick section having a 0.4 design lift coefficient and is just about the same, both smooth and rough, as that of the older NACA 23012 shown here for comparison. The maximum lift coefficient is decreased and the effect of increasing airfoil thickness ratio is less pronounced when the airfoil leading edges are rough. The increments of maximum lift associated with increasing camber, however, appear to be relatively insensitive to airfoil surface condition.

The lift and drag data which have been presented for airfoils both smooth and rough indicate that the aerodynamic characteristics of airplane wings are strongly influenced by surface condition. The predicted characteristics of airfoil sections when applied to airplane wings must, therefore, include careful consideration of the tolerances to which the wing is to be manufactured and the conditions under which the airplane is expected to operate. In any case, however, the characteristics of low-drag airfoils are no worse than those of conventional airfoils and, if sufficient care is taken with the surface condition, definite advantages are associated with their use.

The problem of developing good high lift devices for various airfoils was the subject of considerable research before the war with the older types of airfoils and during the past few years this work has been continued with the low-drag airfoils. Each of the airfoils in the systematic series was tested with a 20-percent-chord split flap deflected 60°. These data show that, although the highest maximum lifts for plain airfoils were obtained for thickness ratios of approximately 12 percent, the maximum lift coefficients of the airfoils with split flaps increased up to thicknesses of 18 or 20 percent. Maximum lift coefficients as high as 2.80 have been measured in this thickness range.

Section maximum lift coefficients are shown in figure 4 for low-drag airfoils of 16-percent thickness equipped with plain, split, and slotted flaps and for a 12-percent-thick low-drag airfoil with a double slotted flap. With a 15-percent-chord leading-edge slat and a single boundary-layer suction slat located at 40-percent chord, the airfoil with double slotted flap had a maximum lift of 3.9. Some data are available which indicate that maximum lift coefficients well over 4.0 can probably be obtained with these high-lift devices on airfoils of 18-percent thickness.

It is perhaps of interest that with flaps deflected the drag coefficients obtained for airfoils equipped with slotted flaps are much lower than those obtained with split flaps, indicating that the slotted flap is much better from the standpoint of short take-off run.

The results obtained for airfoils with split flaps as well as some data for slotted flaps show that the decrement in maximum lift coefficient caused by leading-edge roughness is the same for airfoils both with and without flaps.

A complete presentation and analysis of aerodynamic data for the 100 related low-drag airfoils as well as new two-dimensional data for the 4- and 5-digit series airfoils are now available in reference 2. With the use of this report, the airfoil best suited for a given airplane may be selected. Some of the data included in the summary may be selected. Some of the data included in the summary airfoil report are now being extended to lower Reynolds numbers for use by the designer of small airplanes having low landing speeds.

SYMBOLS

c_d section drag coefficient (d/qc)

c_l section lift coefficient (l/qc)

d section drag, pounds

q dynamic pressure, pounds per square foot

c airfoil chord, feet

l section lift, pounds

t airfoil thickness, feet

Subscript:

max maximum

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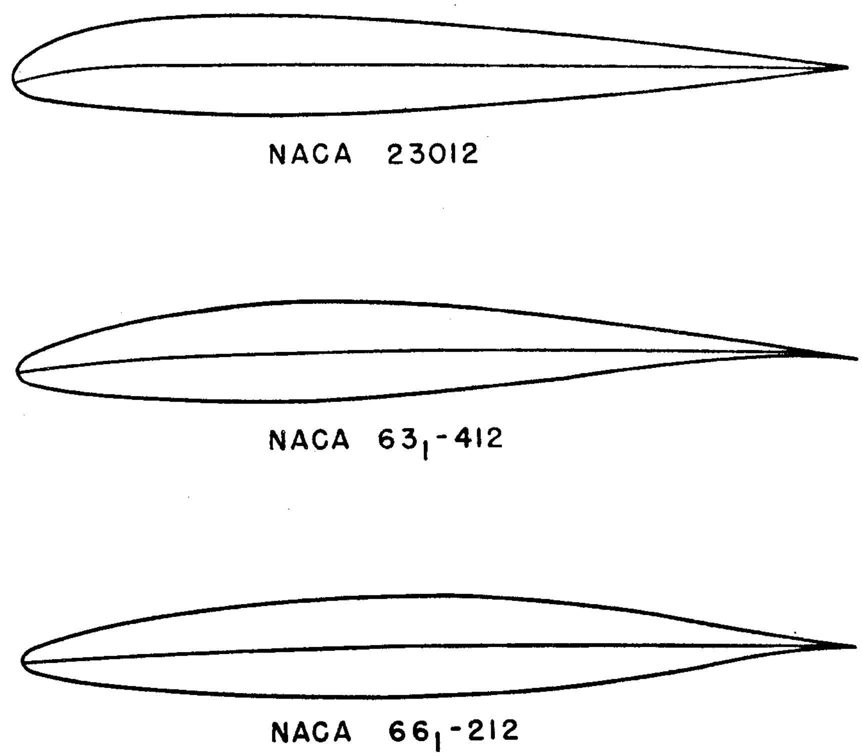

Figure 1.- Comparison of shape of two NACA low-drag airfoil sections with the NACA 23012 airfoil section.

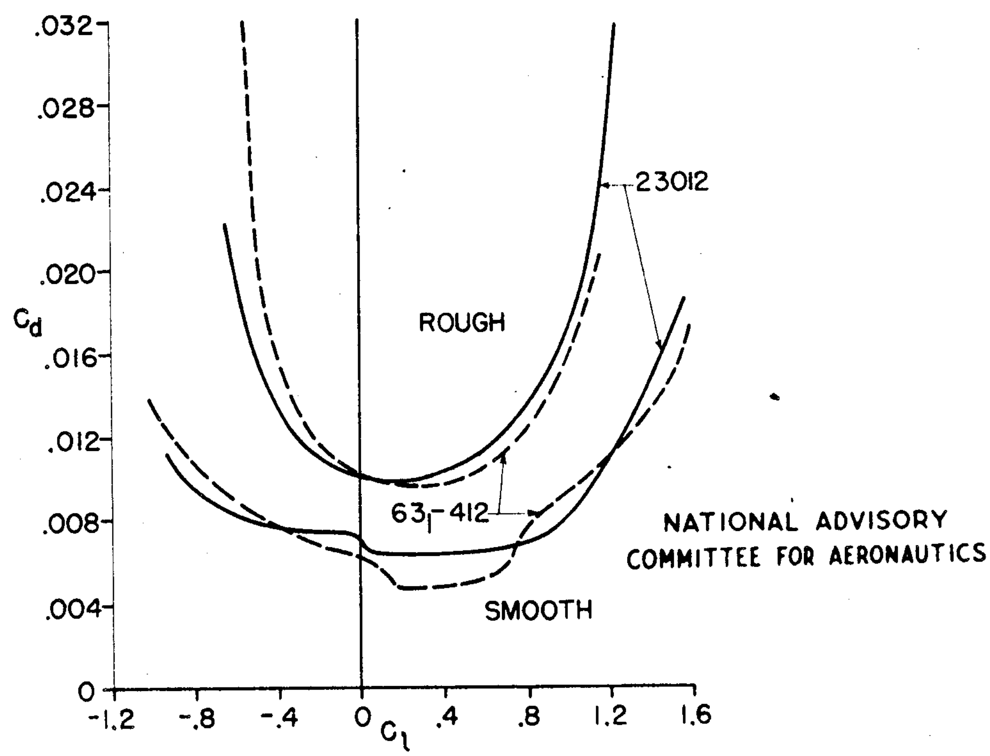

Figure 2.- Drag characteristics of NACA low-drag and conventional airfoils with both smooth and rough leading edges; R = 6 × 10^6.

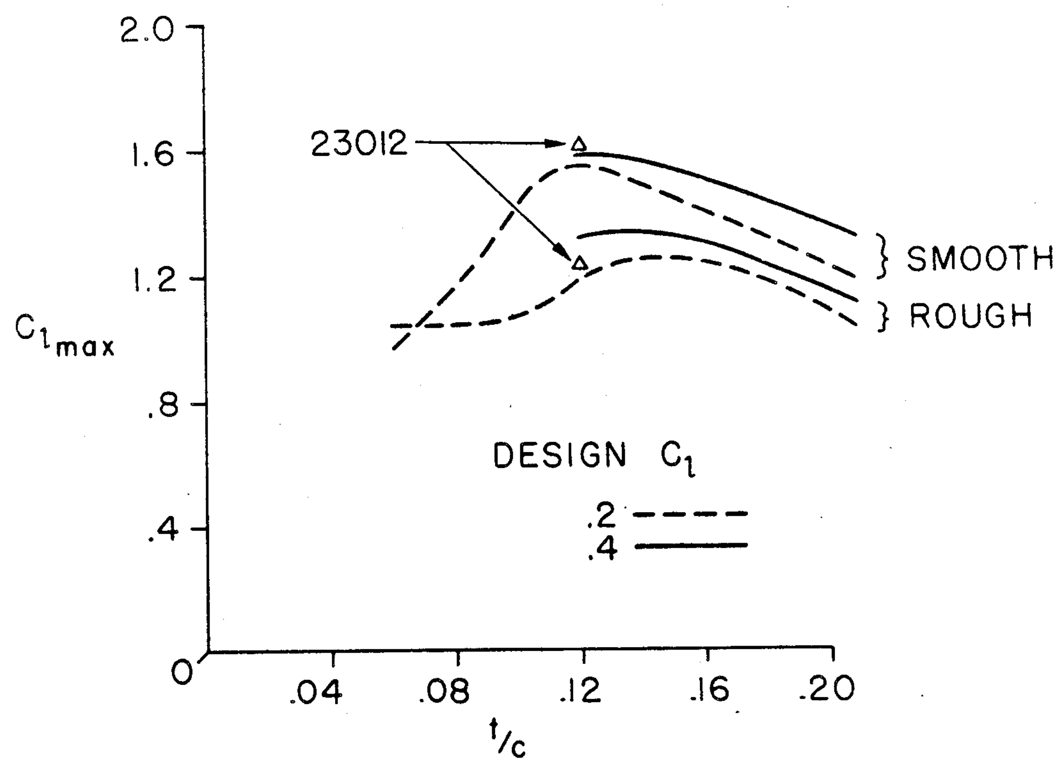

Figure 3.- Variation of maximum section lift coefficient with airfoil thickness ratio and camber for NACA 63-series low-drag airfoils having both smooth and rough leading edges; $R = 3 \times 10^6$.

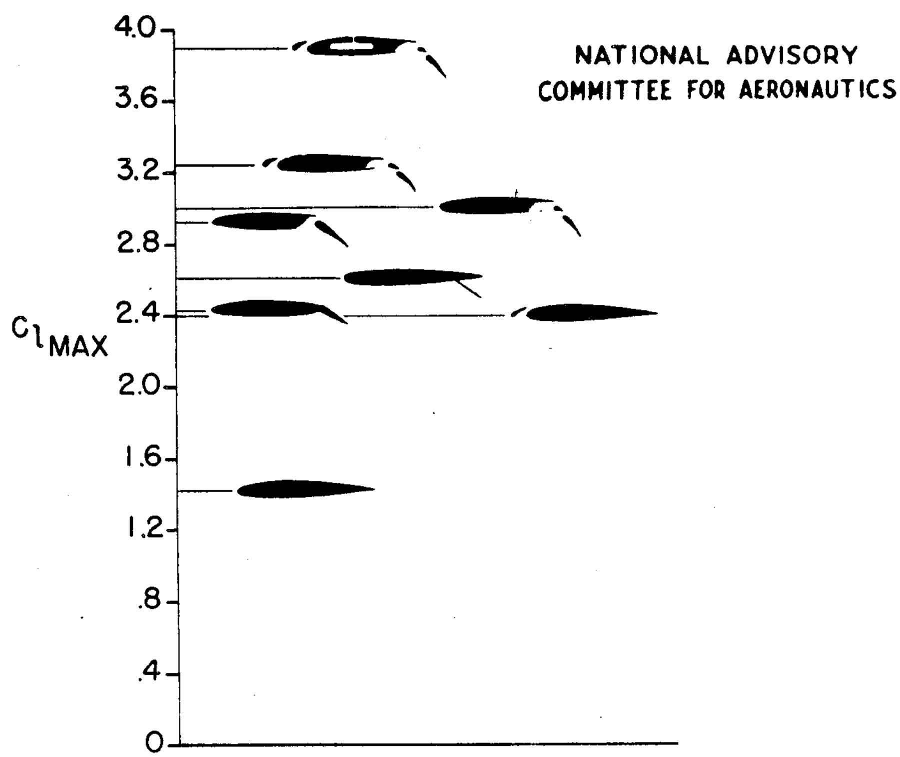

Figure 4.- Maximum section lift coefficients obtained with NACA low-drag airfoils having various types of high-lift devices.

MAXIMUM LIFT AND STALLING

By Harold H. Swoberg

The problem of predicting the maximum lift coefficient and the point of initial stalling of a wing from available two-dimensional airfoil section characteristics can be most readily solved by the use of lifting-line theory. Solutions to this problem have been presented by numerous writers using linear section lift data. At high angles of attack, however, the section lift curves are not linear and hence more refined methods of calculation are necessary. A report, now in the process of publication (reference 1), presents a method for calculating the lift distributions and the force and moment characteristics of wings using nonlinear section-lift data. A comparison is given in figure 1 of calculated and experimental lift results for a wing of aspect ratio 8 and taper ratio 2.5. The calculations are given using both nonlinear and linear section-lift data. The lift curves calculated using nonlinear section-lift data are in close agreement with the experimental results over the entire range of lift coefficients, while those calculated using linear section-lift data are in agreement only over the linear portion of the curve, as would be expected. It should be remembered that the methods presented are subject to the limitations of lifting line theory. The close agreement shown in figure 1 should not be expected for wings of low aspect ratio or large sweep.

The point of initial stalling of a wing can be determined from curves showing the distribution of section lift coefficient along the span together with the distribution of the maximum lift coefficients of the individual sections. An example of calculations to determine the point of initial stalling and the possible rate of stall progression for the same wing described in figure 1 is given in figure 2 together with some experimental results. Stalling should begin at the points of tangency of the section lift curve and the section maximum lift curve. The rate at which the section lift and section maximum lift curves separate is believed to be an indication of the possible rate of stall progression. The difference between the curves is the margin between the actual lift coefficient and the stalling lift coefficient of the sections. When the margin is small, a slight disturbance may produce a stall over a large portion of the wing. The calculations indicate separation over the central and inboard portions of the wing semispan with no evidence of any tip-stalling tendencies. These facts are substantiated by the results of tuft observations shown in the upper part of figure 2. Although the distributions of figure 2 show a small margin between the section lift and section maximum lift curves except at the wing tip, possibly indicating a more rapid spread of separation coupled with a sharper drop in lift past $C_{L_{max}}$ than are shown in figures 1 and 2, it should be noted that the calculations were made for a wing employing 44-series sections which possess gradual stalling characteristics. A more rapid spread of separation, and a sharper drop in lift past $C_{L_{max}}$ would probably have been measured if sections having abrupt stalls, say, 230-series sections, were used.

In the preceding discussion, it was shown that methods are available for predicting with sufficient engineering accuracy the maximum lift coefficient and the point of initial stalling of the wing alone. The question now arises as to how the results of these calculations are modified by the addition of a fuselage or nacelles to a wing, by wing-surface roughness and air leakage through a wing, by partial-span flap deflection, and by propeller operation.

The addition of a fuselage or nacelles to a wing will generally introduce small local areas of separation near the wing center section. This effect may tend to improve the handling characteristics of an airplane near the stall inasmuch as the wake from the stalled regions may cause tail buffeting or a loss of control effectiveness, thus providing a stall warning. Furthermore, flow separation from the inner parts of the wing will generally cause a nose-down pitching moment and thereby increase the longitudinal stability near the stall. The loss in maximum lift coefficient caused by small areas of separation around the fuselage and nacelles will generally be compensated by the added lift on the fuselage and nacelles unless exceptionally poor wing-fuselage and wing-nacelle junctures are incorporated on the airplane.

Wing surface roughness and air leakage through a wing should be avoided wherever possible inasmuch as their effects on maximum lift are detrimental. Full-scale wind-tunnel tests of two airplanes showed that increases of between 10 and 15 percent in the maximum lift coefficients of these airplanes were obtained when all protuberances were faired and when all points of air leakage on the wings were sealed.

Analyses of wind-tunnel and flight data relating to the changes in the stalling characteristics of airplanes resulting from partial-span flap deflection have shown that the results may be divided into four main effects:

(1) An increase in upwash and effective angle of attack over the outer unflapped parts of the wings thereby increasing the tendency for tip stalling.

(2) A tendency to delay separation over the inner parts of the wings.

(3) A tendency to cause a more sudden loss in lift at the stall than in usually measured for the unflapped wing, and

(4) An increase in stall warning provided the flap wake envelops the tail at moderate to high angles of attack.

The first three of these effects tend to produce poor stalling characteristics while the fourth effect tends to improve the handling characteristics of an airplane near the stall. Full-scale-tunnel measurements showing the effects of partial-span flap deflection on the stall progressions of three airplanes having untwisted wings with widely different plan forms is given in figure 3. One airplane has a highly tapered wing (taper ratio, 4:1), another has a wing with low taper ratio (taper ratio, 1.5:1), and the third airplane has a wing with elliptical chord distribution. The smaller angles of attack for each stall progression shown in figure 3 correspond roughly to the angle for initial stalling while the higher angles of attack correspond roughly to the angle for maximum lift. With flaps retracted, the stall progression for each wing is characteristic of its particular plan form. For the wing with high taper ratio, with the exception of small localized areas of separation behind the nacelles and at the wing-fuselage junctures, the main stall originates at the wing tips and progresses inboard with increasing angle of attack. For the wing with low taper ratio the stall originates at the inboard parts of the wing and progresses outboard with increasing angle of attack The stall originates about 80 percent of the semispan inboard from the wing tip for the wing with elliptical chord distribution and spreads both inboard and outboard with increasing angle of attack. For all three airplanes, flap deflection tended to delay separation over the inboard portions of the wings. No small areas of separation appeared at the wing trailing edge near the root section of the highly tapered wing, and for the other two airplanes, initial stalling occurred at a higher angle of attack with flaps deflected than with flaps retracted. The lift curves corresponding to the stall progressions of the three airplanes noted in figure 3 showed that, in each case, a more abrupt loss in lift past the maximum lift coefficient was measured with flaps deflected than with flaps retracted. This effect was very pronounced in the case of the airplane with wing of elliptical chord distribution. It is worth mentioning that flight tests showed this airplane to have exceptionally poor stalling characteristics.

The effects of propeller operation on the stalling characteristics of an airplane can be illustrated by the full-scale-tunnel measurements shown in figure 4. With the propeller removed, the stall progression with angle of attack is typical of that for a low-taper-ratio wing and is fairly symmetrical on both sides of the plane of symmetry. With the propeller operating at a thrust coefficient of 0.2, which corresponds to a power-on landing with about 1/4-rated power applied, the wing sections behind the upgoing propeller blades stalled at a considerably lower angle of attack than the wing sections behind the downgoing propeller blades. An asymmetrical stall pattern is thus produced. Measurements of the variation of rolling-moment coefficient with angle of attack showed that, with the propeller removed, the airplane rolling-moment coefficient was essentially independent of angle of attack (see fig. 4); whereas, with the propeller operating at a thrust coefficient of 0.2, the rolling-moment coefficient increased gradually with angle of attack up to the angle for maximum lift. Above this angle, a sharp increase in the rolling-moment coefficient occurred, which would be sufficient to cause rolling instability in landing. Flight measurements showed that this airplane developed a serious left-wing dropping tendency during power-on landings.

Lateral instability and the consequent wing-dropping tendency is perhaps the most serious feature of the stall. This feature, which is mainly caused by wing tip and asymmetric stalling, can generally be avoided by considerations in the preliminary design stages of an airplane. The use of airfoil sections at the wing tips of moderate thickness (above about 12 percent) and of fairly high camber (about 3 or 4 percent) have been shown to improve the behavior of airplanes having bad tip-stalling tendencies. The use of thin sections (9 percent and below) at the wing tips should be avoided wherever possible especially on wings having taper ratios greater than about 2:1. Both theory and experiment have shown that high taper is conducive to tip stalling. Owing to the loss in aileron effectiveness and damping in roll usually associated with wing-tip stall, several methods have been devised for moving the location of the initial stall inboard. These methods, which include washout, increasing camber from root to tip, central sharp leading edges, and leading-edge tip slats are discussed in detail in reference 2. Sharp leading edges have also been used to induce separation over only one wing panel to lessen the consequences of severe asymmetrical stalling. It should be noted, however, that the use of sharp leading edges will generally cause a loss in maximum lift.

Experimental data have shown the leading-edge slat to be an effective means for reducing the tendency for uncontrolled motions of an airplane in a stall. Flight tests were made on one airplane to investigate the effect of the spanwise extent of a leading-edge slat. The results are presented in figure 5 as time histories of the stall progression and rolling velocity during the stall. Three conditions were investigated:

(1) Airplane without slats

(2) Airplane with slats extending over 53.5 percent of the semispan

(3) Airplane with slats extending over 75.8 percent of the semispan.

With no slats, stalling progressed very rapidly over the outboard portions of the wing and the airplane rolled rapidly to the right with no evidence of any righting tendency. With a leading-edge slat extending over 53.5 percent of the semispan the airplane initially rolled to the right but the roll was halted by the slat so that the airplane remained in a banked attitude. Increasing the extent of the slat to 75.8 percent of the semispan gave the airplane the ability to retain a laterally level attitude throughout the stall; however, the airplane exhibited objectionable longitudinal oscillations which were not measured for the other two conditions. It is concluded from these results that, for this airplane, the optimum extent of leading-edge slat would be about 65 percent of the semispan. These and other data indicate that the large slat span required, together with the disadvantages of increased drag if the slat is fixed and added mechanical complications if the slat is retractable appear to limit the use of the leading-edge slat as a means for obtaining satisfactory stalling characteristics.

In conclusion, I would like to reiterate that methods are available for calculating the effects of the wing geometry and of the spanwise lift distribution on the maximum lift and the point of initial stalling of a given wing. These calculated wing characteristics may be radically changed, however, by wing-fuselage and wing-nacelle interference, by partial-span flap deflection, by propeller operation, and by poor wing-surface conditions. The extent of these effects can be most readily evaluated from analyses of the available data on maximum lift and stalling research. In this connection a summary of available full-scale-tunnel data relating to maximum lift and stalling research has recently been made available for general use (reference 3). Finally, it is felt that sufficient experimental data are now available to show that by carefully selecting the proper combinations of wing geometry with washout and with changes in airfoil camber and airfoil thickness from root to tip satisfactory airplane stalling characteristics may be obtained without the use of auxiliary devices such as central sharp leading edges and leading-edge slats.

SYMBOLS

C_L lift coefficient $\left(\frac{\text{Lift}}{qS}\right)$

q dynamic pressure, pounds per square foot

S wing area, square feet

α angle of attack, degrees

R Reynolds number

c_l section lift coefficient

C_l rolling-moment coefficient $\left(\frac{L}{qbS}\right)$

L rolling moment, foot-pounds

b wing span, feet

T_c effective thrust coefficient $\left(\frac{T}{\rho V^2 D^2}\right)$

T thrust, pounds

ρ density, slugs per cubic foot

V airspeed, feet per second

D propeller diameter, feet

Subscript:

max maximum

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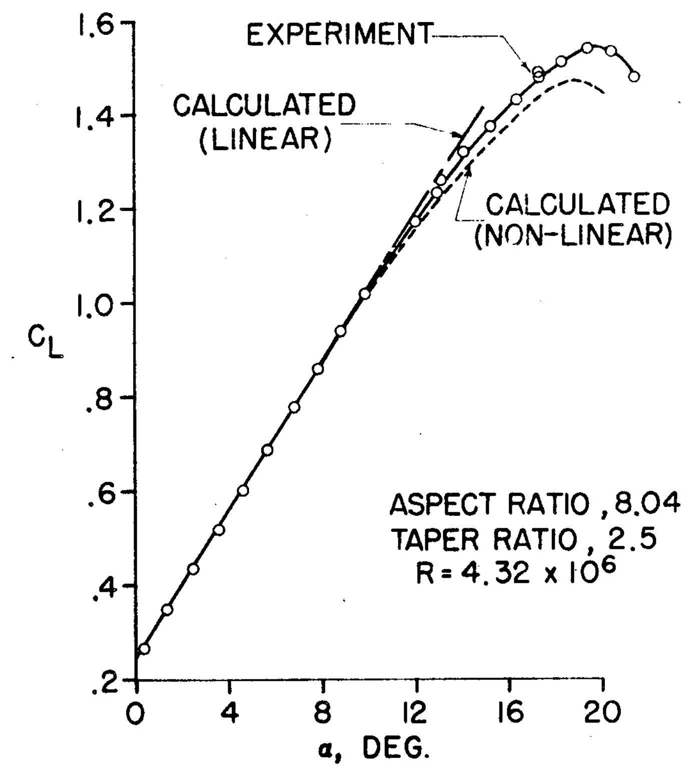

Figure 1.- Comparison of experimental and calculated wing lift characteristics. Root section, NACA 4416; tip section, NACA 4412.

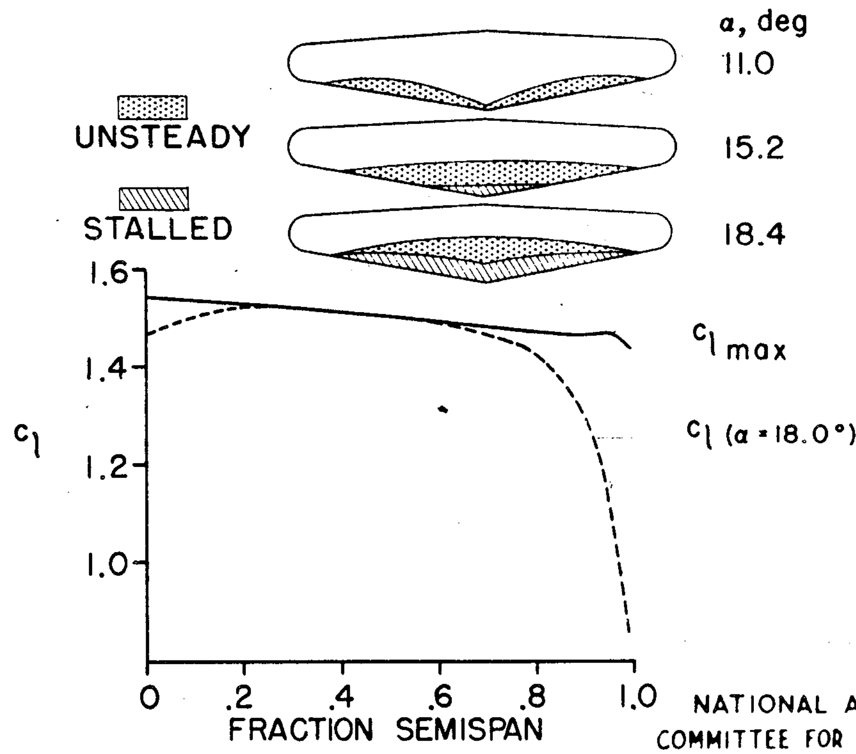

Figure 2.- Calculated and experimental stalling characteristics for wing described in figure 1.

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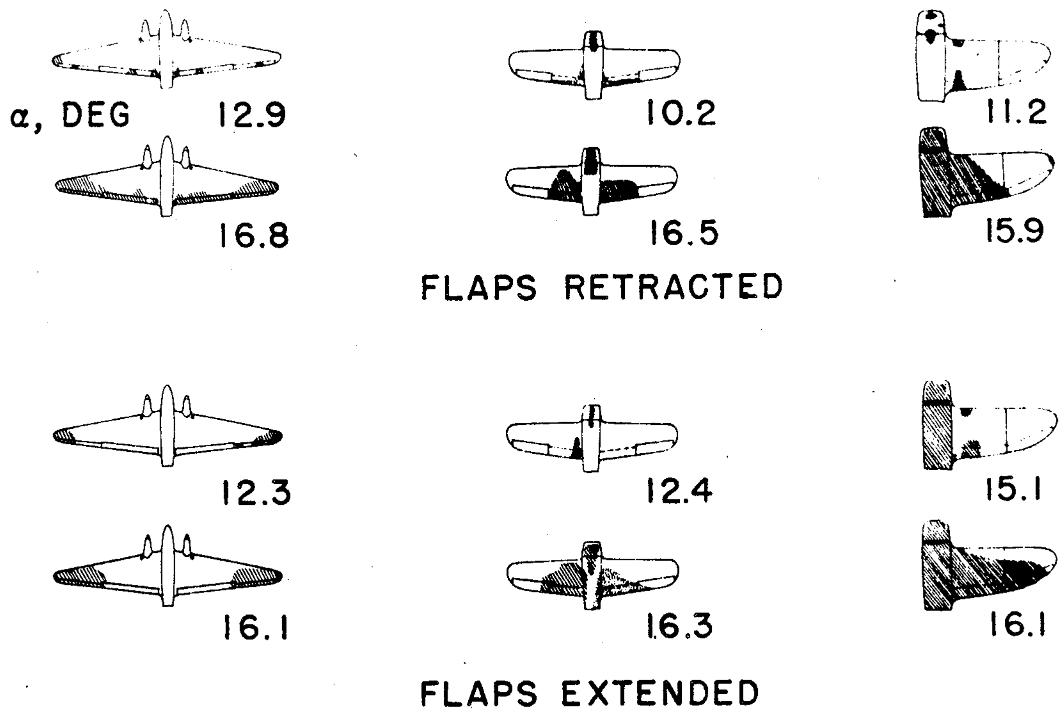

Figure 3.- Effect of flap deflection on stall progression of airplanes having untwisted wings of different plan form. Airplanes tested in the Langley full-scale tunnel.

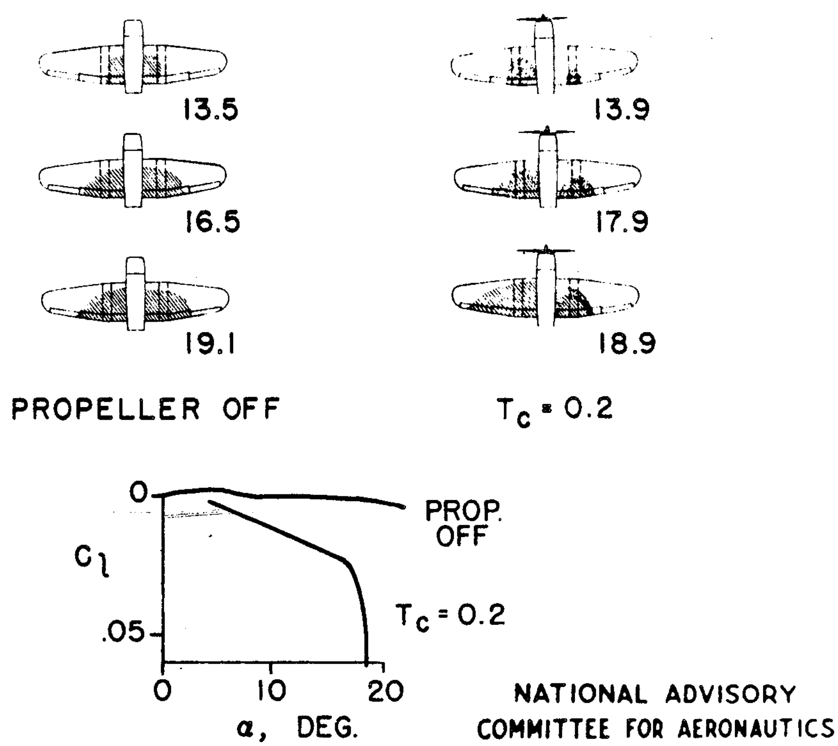

Figure 4.- Full-scale tunnel measurements showing the effect of propeller operation on stalling characteristics.

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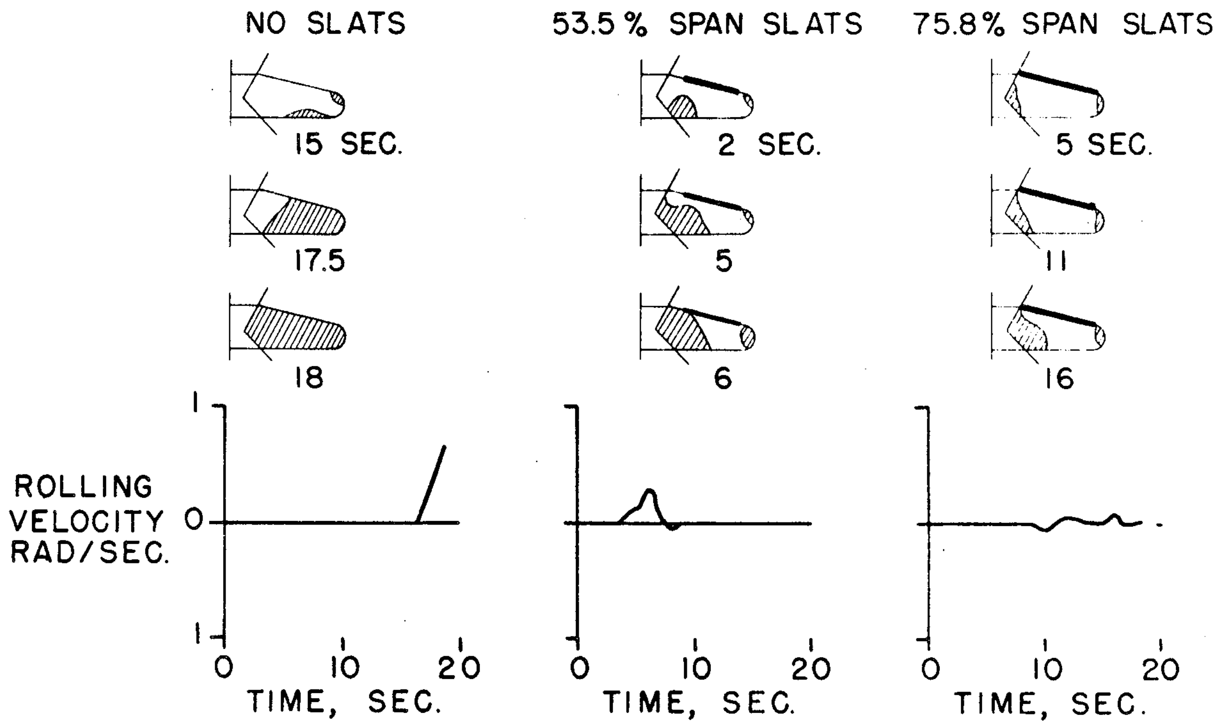

Figure 5.- Measurements in flight to investigate the effect on stalling behavior of spanwise extent of leading-edge slat; flaps extended and throttle closed.

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PROPELLERS

PROPELLER SELECTION AND EFFICIENCY

By John L. Crigler

A method of selecting a propeller for a given airplane installation based upon theoretical calculations and experimental data has been developed. This method can be used to design an optimum propeller for any operating condition and will accurately predict the propeller performance. The method provides a breakdown of the propeller power losses and thus allows the best compromise of these losses to be made throughout the entire operating range.

Experimental verifications of this method for specific applications for which test data are available have been obtained over the entire operating range. Publications on the methods of theoretical calculations and on experimental verifications are available. (See references 1 to 6.)

Several charts have been prepared to illustrate the breakdown of propeller power losses for a range of operating conditions.

Figure 1 shows the variation of the ideal propeller efficiency with speed for propellers absorbing 100 horsepower. This figure covers the take-off speed range only. The three curves represent three propeller diameters, 6 feet, 12 feet, and 20 feet. This ideal efficiency is calculated for minimum axial energy losses alone and neglects the drag of the blade sections as well as the rotational velocity in the slipstream. It is seen that the propeller diameter fixes the maximum possible efficiency that can be obtained in the take-off range. If a 6-foot-diameter propeller is to absorb 100 horsepower at 30 miles per hour, this maximum possible efficiency is only 48 percent, while a 12-foot-diameter propeller would increase the maximum possible efficiency to 65 percent and a 20-foot propeller to about 80 percent. Although considerations of ideal efficiency favor large diameters in the take-off range, the propeller diameter is, of course, limited by practical considerations such as propeller weight and clearance. Serious losses of efficiency at take-off speeds are therefore unavoidable with practical propellers.

Let us consider next the condition at flying speeds. In this case we will consider a 6-foot-diameter propeller as typical of present practice. Figure 2 shows the available thrust horsepower for operation with the propeller mounted on a 100-horsepower engine. The upper curve in the figure (the solid line) is the same as given in figure 1 but has been extended to higher speeds. Since a 100-horsepower engine was used, the ideal efficiency and the ideal in the figure, shown dashed, shows the calculated thrust horsepower for operation with a 6-foot-diameter, two-blade propeller with Clark-Y sections and round blade shanks. The propeller blade has an activity factor of 90 and the solidity at the 0.7 radius is 0.0345 per blade. This curve is for variable pitch operation having an optimum load distribution at 120 miles per hour and is for a constant brake horsepower of 100 over the entire speed range. The difference in power losses between this curve and the curve for the 6-foot propeller with ideal thrust power is divided into three parts; the loss due to the profile drag of the blade sections, including the round blade shanks, the rotational energy loss, and an additional axial energy loss because the propeller loading is not the ideal as given by the momentum theory. The increase in axial energy is caused by the decrease in load on the propeller hub and tip sections in order to give minimum energy losses. If the load on the hub and tip sections were maintained the increase in rotational energy and the tip losses would be greater than the saving of the axial energy.

The dashed line of the two lower curves gives the thrust horsepower for operation with the same propeller as the second curve but with fixed pitch. Since the fixed pitch propeller would absorb more power at constant rpm as the airspeed is decreased the engine rpm decreases, resulting in a decrease in the brake horsepower output. The major difference between these two curves is the decrease in the brake horsepower. Small differences in the profile drag of the blade, and axial and rotational energy losses, are present but the total difference in propeller efficiency is of the order of 1 to 2 percent.

Calculations were made on the fixed pitch propeller to determine the effects of airfoil section, section thickness, pitch distribution, plan form, and solidity on the propeller efficiency over the speed range. Changes in any of these variables only caused a change in the thrust horsepower of approximately ±1 percent over the range of speeds. The dotted line which was calculated for a similar blade but with a 50 percent increase in solidity is typical of the effect of any one of these changes. The thrust horsepower is about 1 percent lower for the design condition, due to increased profile drag of the blade. In the low-speed range the propeller efficiency for the extra solidity propeller increased by the order of 5 to 10 percent but the thrust horsepower developed by the propeller was only slightly increased because the brake horsepower of the engine decreased. The effect of changes in plan form and pitch distribution may be treated as one problem since both are changes in the propeller load distribution. Neither showed any appreciable change in the thrust horsepower over the range from 50 miles per hour to 130 miles per hour unless radical changes in the propeller were made. Extreme changes would result in a further reduction in the thrust horsepower. It is thus seen that of all these variables, variable-pitch operation is the only change that gives any appreciable improvement in thrust horsepower in the take-off range.

In the development of the propeller theory the induced losses are treated separately from the drag losses and a propeller may be considered as two separate propellers, one producing lift alone and the other producing drag alone. Figure 3 shows the element efficiency loss due to the profile drag of the blade sections plotted against $\frac{V/nD}{x}$ for a range of values of the ratio of lift to profile drag, which is the drag used in propeller calculations. Typical sections operating at lift coefficients of about 0.55 have a lift-drag ratio of approximately 70. From the figure the importance of the drag loss for any operating condition can be determined. The total power loss due to drag is about 3 percent for sections operating at a lift-drag ratio of 70 in the best operating range of V/nD and small changes in the section lift-drag ratio due to changes in the airfoil or section thickness are relatively unimportant. In the take-off range the drag loss increases to about 8 percent of the total power but here again it is seen that small changes in the section lift-drag ratios result in small changes in the drag loss. A change of ±10 percent in the lift-drag ratio of the sections results in a change of approximately 1 percent in the propeller efficiency. It may be concluded that no substantial gain in propeller efficiency for light airplanes can be realized by further research to obtain improved propeller sections.

We shall now consider briefly the effect of body interference on the propeller efficiency, since the propeller usually operates in a field of nonuniform axial velocity. If this nonuniformity is constant around the circumference, such as is the case when the propeller is mounted in front of a blunt body such as an NACA cowling and the velocity distribution is known the effect may be taken into account in the propeller design. In the calculations the change in velocity from the free-stream velocity simply results in a change in the effective V/nD of the propeller section. It is important that this effect be included in the propeller design as an optimum designed propeller for free-stream operation is no longer optimum when placed in a field of nonuniform axial velocity. For example, suppose a propeller that is designed to operate at a given operating condition is so mounted in front of a blunt body that the inner third of the radius of the propeller operated in a low velocity region. If good airfoil sections extended in close to the hub these inner sections would be overloaded with a resultant decrease in propeller efficiency. This shift of the load distribution is not serious, however, for low V/nD operation unless it is enough to stall some of the sections or unless the velocity distribution is such that some sections are completely unloaded. If the airfoil characteristics are known and the velocity distribution in the propeller plane is known the resultant propeller efficiency can be very accurately computed for operation in a field of nonuniform axial velocity, provided this nonuniformity is constant around the circumference, and the propeller can be made optimum just as well as for free-stream operation.

If the nonuniformity in axial velocity cuts across the propeller disk as in the case of a propeller mounted close up behind a wing the effect cannot be included in the calculations. It is difficult to predict what happens to the propeller efficiency and tests are always required to determine the efficiency for a particular location. For instance, if the propeller were designed to operate in the region of high velocity it would stall if operated at the same rpm and pitch setting in a low velocity region. Whether the propeller can carry the load alternately through the regions of high and low velocity without stalling is a matter that can only be determined by tests on the particular arrangement.

Figure 4 shows results of tests that were conducted in the NACA propeller-research tunnel to determine the propulsive efficiency of a propeller operating behind a wing with a deflected flap. A general arrangement of the test model is shown. The model used was a constant-chord wing with nacelle and single-slotted flap duplicating the arrangement at the center nacelle of a high-speed bomber.

Modifications to the original model consisted of the installation of a split flap.

Tests were made over a range of blade angles, flap deflections, and angles of attack to be encountered in take-off and climb.

Although the propeller was mounted close up behind a large nacelle it should be noted that the propulsive efficiency is very good for the 0° flap setting; 85 percent at a V/nD of 0.80. Deflecting the single-slotted flap to 40° reduced the efficiency by about 8 percent at $V/nD = 0.77$ and power coefficient, $C_p = 0.080$. For the same conditions the split flap reduced the efficiency by about 23 percent.

This loss in efficiency could be caused by flaps, landing gears or any body that causes nonuniformity of axial air flow. The same type of flow with the resulting loss can occur if the propeller is mounted too close in front of a wing but the velocity gradient with distance is much sharper in front of the wing than behind so that there is less likelihood of getting into trouble unless the propeller is mounted very close to the leading edge of the wing.

When such conditions exist and the blade stalls on passing through the low velocity region the losses in propeller efficiency will be very serious.

SYMBOLS

D	propeller diameter
D	drag of propeller blade element for infinite aspect ratio
L	lift of blade element
L/D	lift-drag ratio
R	propeller tip radius
r	radius to any element
x	radial location of blade element (r/R)
n	propeller rotational speed
V	axial velocity of propeller
V/nD	propeller advance-diameter ratio
η	propeller or element efficiency
$\Delta\eta_D$	element efficiency loss due to drag

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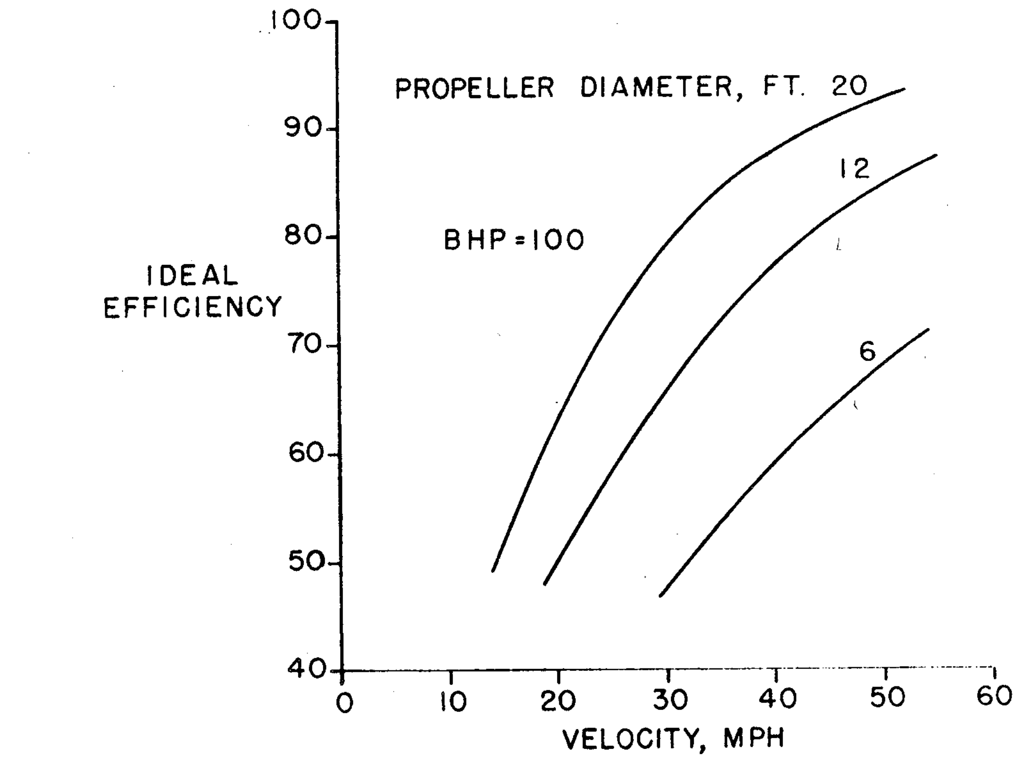

Figure 1.- Variation of ideal propeller efficiency with velocity.

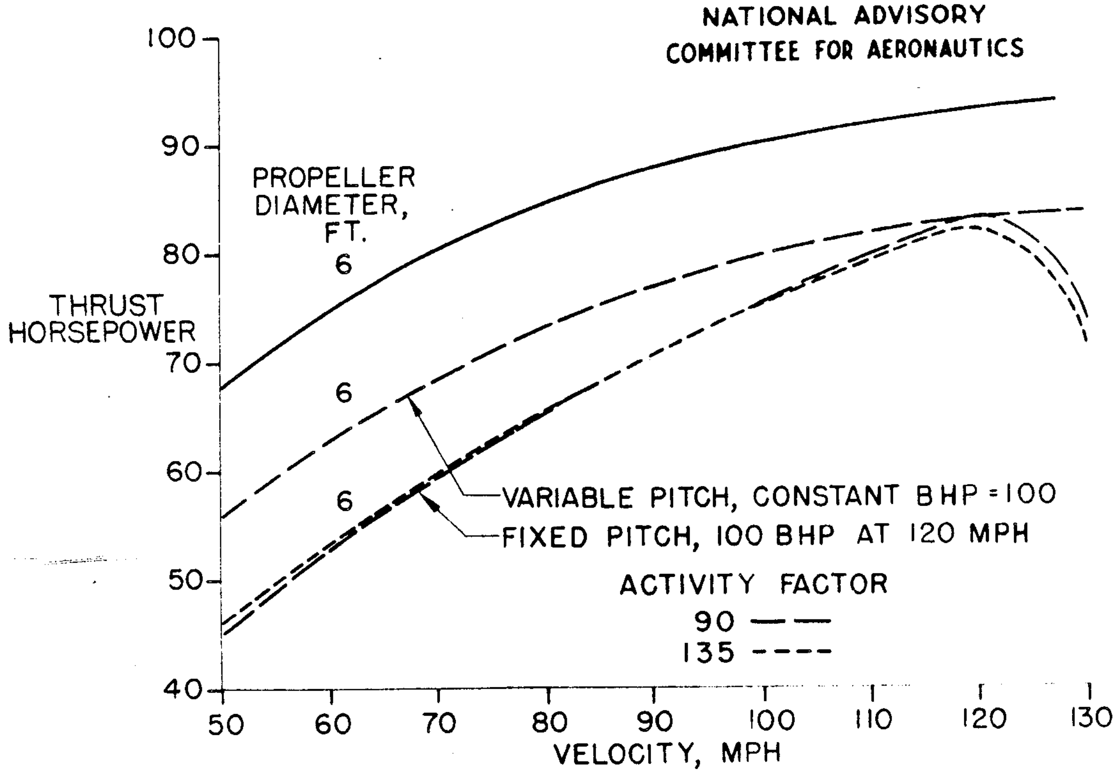

Figure 2.- Variation of thrust horsepower with velocity for a 6-foot-diameter propeller.

Crigler

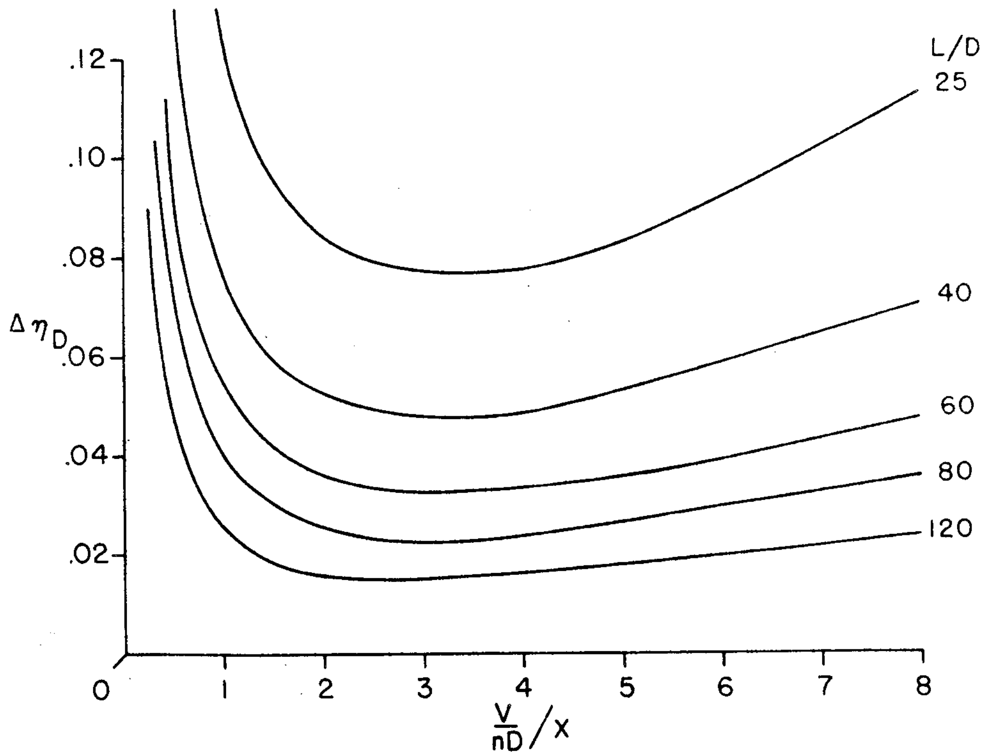

Figure 3.- Element efficiency loss due to drag for a range of values of L/D.

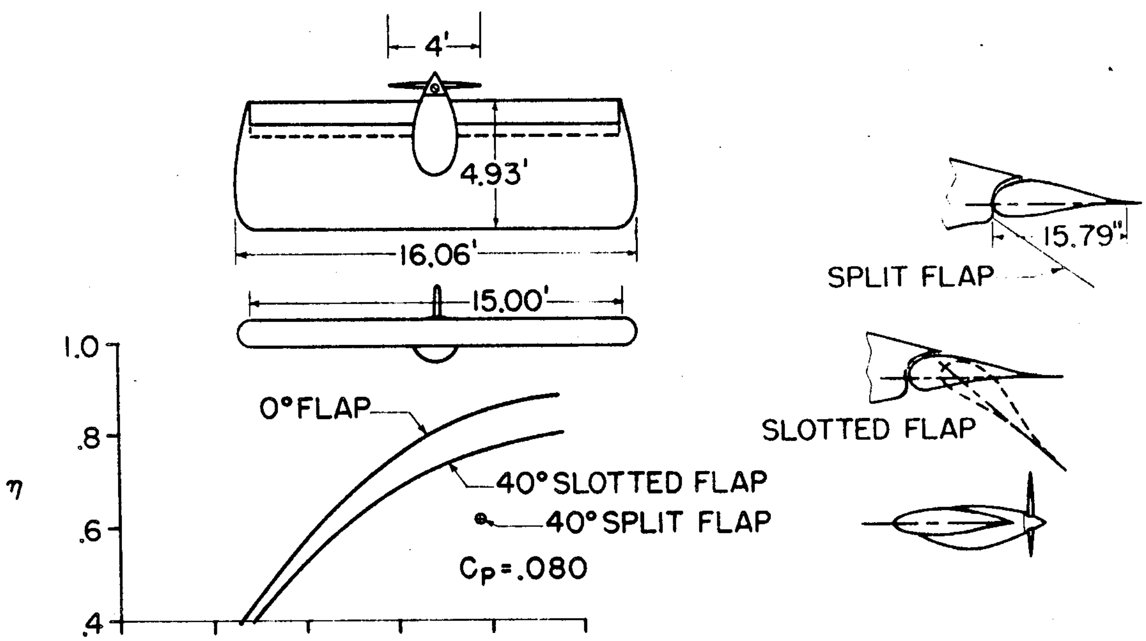

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Figure 4.- Effect of flap deflection on propulsive efficiency at constant take-off power.

PROPELLER NOISE

By Arthur A. Regier

INTRODUCTION

The problem of airplane noise is not a new one here at the Langley laboratory of the NACA. Work on this problem was started about 16 years ago. It was discontinued about 8 years ago because a general theory of propeller noise had been found and confirmed. Some of this work is reported in reference 1. The sound theory and tests showed that in order to eliminate propeller noise, or materially reduce it, a radical redesign of the propeller and possibly the airplane was necessary. At that time, the chief interest of the aviation industry was in high performance military airplanes and it was felt that such redesigns were not practical for military airplanes.

Recently, however, work on propeller noise has been resumed at the request of Dr. T. P. Wright, CAA Administrator. This work has been concerned primarily with the noise of light airplanes as heard by people living in the vicinity of small airports. The CAA is interested in this problem because it may be the determining factor in locating the new airports provided for in the new Federal Airport Expansion program. Also, the elimination of airplane noise will greatly increase the comfort of the passengers of the airplane.

A general review of the problem of airplane noise with special reference to light airplanes is given in a recent NACA publication, reference 2. It is pointed out that the predominating source of noise in present day aircraft is the propeller and that silencing the engine is useless unless something is done about the propeller. When propellers are silenced it is believed that engines can also be silenced by conventional methods. The present discussion will therefore be restricted to propeller noise.

THEORY OF PROPELLER NOISE

The propeller noise theory was developed in Russia by Gutin. His hypothesis is simply that the steady aerodynamic forces on the blade (thrust and torque) are imparted to the air and propagated through space at the speed of sound in conformity with the classic laws for sound propagation. In order to simplify the mathematics, Gutin considers a small section of air in the propeller disk at a representative radius. He assumes that no forces act on the air until the blade reaches it. At that time the air receives a pressure equal to the pressure on the blade for the time that the blade passes through the section of air under consideration. The air particle therefore receives a square wave impulse whose magnitude is equal to the thrust of the blade. This square wave is resolved into its Fourier coefficients or harmonics. It is because the air receives a sharp blow when the blade passes it that propeller noise is made up of the fundamental frequency (revolutions per second times number of blades) and all integral harmonics of the fundamental.

The sound pressure for any point at a distance from the propeller is obtained by integrating the sound radiated from all the particles on the circle described by the representative section of the propeller. The basic Gutin theory is concerned only with the steady air forces on the blade. If the forces on the blade are caused to vary by nonuniform flow fields induced by fuselages or wings, the sound output of the propeller is increased above that given by the theory. The sound due to the steady air forces will be designated the Gutin sound in this discussion.

There is another source of propeller noise due to the vortices shed from the propeller. These sources are not steady sources on the blade as are those considered in the Gutin theory, but are oscillatory disturbances. The frequency from these oscillatory vortex disturbances is almost a continuous spectrum ranging from a few hundred cycles per second to several thousand. The sound as observed by the listener is similar to that of wind blowing through a forest or of waves breaking on a beach. The Gutin noise on the other hand has only frequencies which are multiples of the fundamental blade passage frequency. Some work on vortex noise is reported in reference 3.

For normal propellers operating at tip speeds above a Mach number of 0.5 these vortex noises are small compared to the Gutin noise. For low-tip-speed fans, however, the vortex noise may become large compared to the Gutin noise. For example, household fan manufacturers have developed fans in recent years to reduce greatly the sound output. They have achieved this mainly by reducing the vortex noise, which depends on tip load conditions and flow conditions on the blade.

There is no significant clean up to be achieved on propellers until the tip speed has been reduced sufficiently so that the Gutin noise is below the vortex noise level. If the propeller tip speed is reduced sufficiently to bring the Gutin noise down to the vortex level, the propeller will be silenced to about the level of traffic noise.

If there is no dissipation or loss of sound energy in the atmosphere, the sound pressure varies inversely as the distance. The change in sound pressure is given in decibels by the following expression.

$$DB = 20 \log_{10} \frac{d_2}{d_1}$$

where $\frac{d_2}{d_1}$ is the ratio of the distances. Thus if the sound pressure is 100 DB at a distance of 30 feet from the source, the pressure will be 20 DB less or 80 DB at a distance of 300 feet. For large distance account must be taken of the additional reduction of sound pressure due to the dissipation in the atmosphere. This additional loss is a function of the sound frequency and atmospheric humidity and has not been accurately determined. It is known, however, to be small for short distances.

DISCUSSION OF FIGURES

Table I gives the effect of tip speed on the sound of the propeller. In the first line of the table all quantities are given a value of unity without regard to the absolute magnitude of the quantity. In the second line the relative values of the quantities are given for twice the tip speed. It is well known that doubling the speed of a propeller increases the power consumption by a factor of 2^3 or 8. This figure is shown in the second column. The Gutin sound shown in the third column varies as the tenth power of the tip speed;hence,doubling the tip speed increases the sound output by a factor of 2^{10} or 1024. Thus it may be seen that by doubling the tip speed the horsepower to the propeller has increased by a factor of 8, but the sound energy radiated has been increased by a factor of 1024. If the relative Gutin sound is divided by the relative power to the propeller (1024/8 = 128) it is seen that doubling the tip speed causes 128 times as much sound energy to be radiated per horsepower to the propeller. It is because the sound energy varies approximately as the tenth power of the tip speed, that tip speed is the most important factor in reducing propeller noise.

The vortex noise given in column 4 varies as the 5.5 power of the tip speed. The ratio of the Gutin noise to vortex noise is given in the last column. It may be seen that doubling the tip speed increases the ratio of Gutin noise to vortex noise by a factor of 23. Hence the vortex noise can be neglected for the higher tip speeds, but may become the predominate source of noise at very low tip speeds.

In the second line third column it is indicated that the sound energy ratio of 1024 is equal to 30 DB or decibels. The decibel is a term commonly used in sound measurements. It is 10 times the logarithm of the power ratio. Thus the logarithm of 1024 is 3 or 30 decibels. In subsequent figures the decibel sound reading will be referred to the commonly accepted base level of 10^{-16} watts per square centimeter which is the threshold of hearing for the human ear at a frequency of 1000 cycles per second.

Recently some sound measurements have been made by the NACA on 2-, 4-, and 7-blade propellers. These propellers were not designed for this problem, but were used because they were available from some high-speed propeller tests. The purposes of the tests were (1) to determine vortex noise level for propellers of many blades and (2) to determine how the noise from such a propeller sounded to various listeners. Some of the results of these tests are given in figure 1. The solid lines give the experimental values for the 2-blade and the 7-blade propellers. The dashed lines give the theoretical values of the sound level as calculated by the Gutin theory. The agreement between experiment and theory is good for the 2-blade propeller over the entire range of tip Mach numbers from 0.3 to 0.9. There is a constant discrepancy of about 2 decibels which may be due to microphone calibration. The agreement between experiment and theory is also good for the 7-blade propeller at the high tip Mach number but becomes progressively worse at the low Mach numbers. This discrepancy is believed to be due to the vortex noise which is not considered in the Gutin theory. The sound spectrum obtained from the 7-blade propeller confirms this belief. At the high tip speeds the sound is almost a pure note equal to the blade passage frequencies and its harmonics. At the low tip speeds the vortex noise drowns out or masks the characteristic Gutin propeller sound.

It should be emphasized that for a given propeller configuration, tip speed, power, thrust, and torque, the Gutin theory predicts the sound pressure. The good agreement between theory and experiment on the 2-blade propeller indicates that the propeller tested is as good as can be obtained for the particular test conditions of power and tip speed. Further research on this propeller would be useless. It is seen, however, that on the

7-blade propeller operating at low tip speeds the measured values exceeded the minimum values predicted by the Gutin theory. Further research may be useful in educing the vortex noise on this propeller.

The horsepower to the propellers is indicated for both propellers operating at 0.7 tip Mach number. The 2-blade propeller absorbs 59 horsepower and the 7-blade propeller absorbs 97 horsepower. In spite of the fact that the 7-blade propeller is absorbing nearly twice the power, the sound pressure is down by 10 decibels. The reason for this is that the thrust has been distributed over more blades with the result that the load per blade is reduced. This means that the Gutin pulses are less for the 7-blade propeller than for the 2-blade propeller. There is another gain for the 7-blade propeller because only harmonics of the blade passage are propagated. Thus, the 2-blade propeller radiates sound at frequencies of 2, 4, 6, 8, etc. times the propeller rotation speed. The 7-blade propeller radiates frequencies of 7, 14, 21, 28, etc. times propeller rotation speed. Since the sound energy in the higher harmonics is less than for the lower harmonics the total sound pressure is less for the 7-blade propeller. The gain for the 7-blade propeller at high tip speeds is not as great as indicated when the characteristics of the human ear are considered. This subject will be discussed later.

Figure 2 shows two propellers which were tested on an electric motor. It may be noted that the Cub propeller operating at a cruising speed of 2100 rpm is absorbing 40 horsepower and radiating sound energy at a level of 107 decibels as measured at a distance of 30 feet from the propeller. The 7-blade propeller having a smaller diameter is absorbing 48 horsepower at the same rpm and the sound energy radiated is 86 decibels, a reduction of 21 decibels. This is not intended as a practical application but serves to demonstrate that the sound pressure can be greatly reduced by operating at lower tip speed and by using a greater number of blades.

The previous discussion has dealt primarily with the sound pressures without regard to the frequencies or the loudness of the sound as heard by the human ear. The characteristics of the human ear are given in figure 3. This chart is taken from reference 4 and is considered as a standard by accoustical workers. The vertical scale gives the sound intensity or pressure in decibels, the horizontal scale the sound frequency. The solid lines give the loudness values as obtained from listening tests made with a large number of observers. The lowest line gives the threshold of hearing. It is designated zero decibels. The loudness is defined as the loudness of a pure tone of 1000 cycles;

therefore, the loudness levels and the intensity levels coincide at 1000 cycles. If the frequency of the tone is reduced, a greater intensity is required to give the same loudness. Thus, if the frequency is reduced from 1000 cycles to 40 cycles, the intensity must be increased from zero decibels to 60 decibels in order to keep the tone audible. In general, for any sound intensity below 100 decibels the ear is much less sensitive to the lower frequencies; therefore, a greater sound intensity may be tolerated at the low frequencies for a given loudness level.

This fact must be considered in a design of a propeller. If the frequency spectrum is raised, the loudness may be increased even though the sound pressure has been reduced. It has been shown that the sound pressure can be reduced by decreasing tip speed and by increasing the number of blades. For a given propeller, decreasing the tip speed also decreases the frequency of the sound as well as the pressure and therefore an additional decrease in loudness is obtained, because of the characteristics of the human ear. If the sound pressure is reduced by adding blades at a given tip speed and diameter, the loudness may not be reduced much because this increases the frequency, thereby placing the noise in a frequency range in which the ear is more sensitive.

The characteristics of the human ear also explain why helicopters are inaudible at a distance of a few hundred feet. The fundamental sound frequency for a helicopter and the first three harmonics fall in a frequency band between 15 and 60 cycles per second, which is a frequency range approaching the low frequency cut-off for the ear. There is no basic reason why airplanes cannot be made to operate as quietly as helicopters. The airplane requires only about one-tenth of the thrust, therefore the sound pressure for the airplane is much less for a given tip speed. The difficulty lies in bringing the frequency of the airplane propeller down to the range in which the ear is insensitive. In order to make the airplane as quiet as the helicopter, it is not necessary to make the sound frequency the same as a helicopter. It is necessary to make the combination of sound pressure intensity and frequency such that the airplane will be on the same loudness level contour as the helicopter. Thus if a helicopter has a sound intensity of 80 decibels at a given distance at a frequency of 30 cycles per second it will have a loudness of only 40 decibels. If the airplane propeller has a frequency of 100 cycles per second, to have the same loudness requires that the sound pressure be reduced to 60 decibels. This is an over simplification of the problem because propeller noise is not a pure tone. The example serves, however, to illustrate the essential features of the problem.

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1. Deming, Arthur F.: Propeller Rotation Noise Due to Torque and Thrust. NACA TN No. 747, 1940.

2 Theodorsen, Theodore, and Regier, Arthur A.: The Problem of Noise Reduction with Reference to Light Airplanes. NACA TN No. 1145, 1946.

3. Stowell, E. Z., and Deming A. F.: Vortex Noise from Rotating Cylindrical Rods. NACA TN No. 519, 1935.

4. Fletcher, Harvey, and Munson, W. A.: Loudness, Its Definition, Measurement and Calculation. Jour. Acous. Soc. Am., vol. 5, no. 2, Oct. 1933, pp. 82-108.

Table I - Relative sound energy for a two-blade propeller as a function of tip speed.

Tip Speed (T.S)	Power to Propeller $(T.S)^3$	Sound Energy Radiated (Gutin) $(T.S)^{10}$	Sound Energy Radiated Vortex Noise $(T.S)^{5.5}$	$\frac{\text{Gutin Sound Energy}}{\text{Power to Propeller}}$	$\frac{\text{Gutin Sound Energy}}{\text{Vortex Sound Energy}}$
1	1	1	1	1	1
2	8	1024 (30db) diff.	45	$\boxed{128}$	23

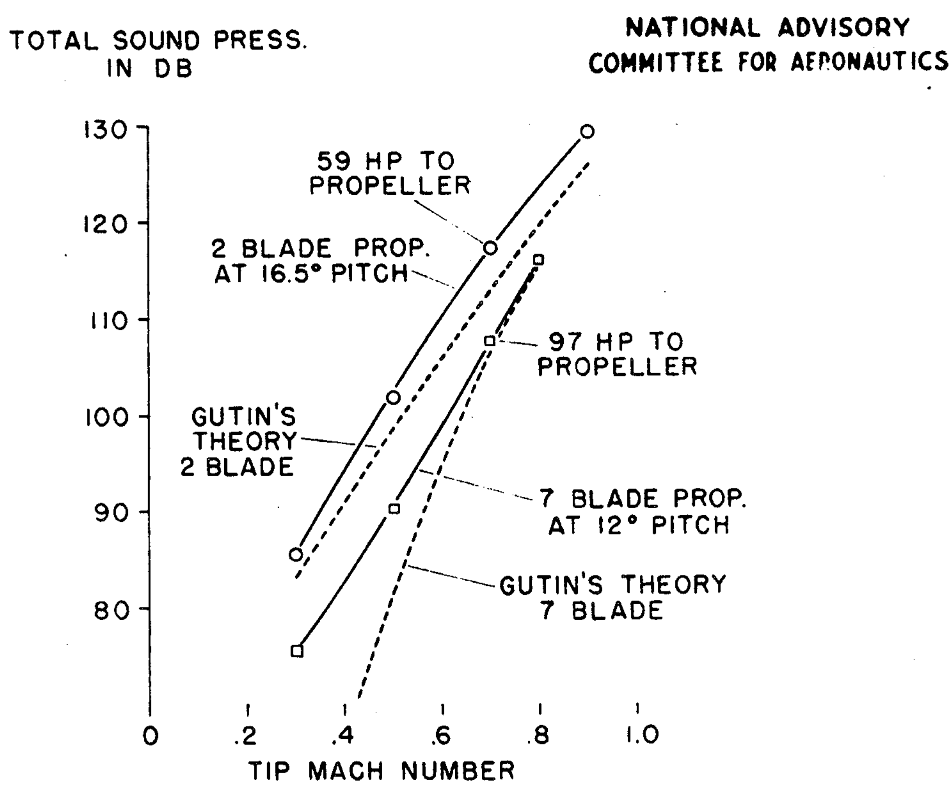
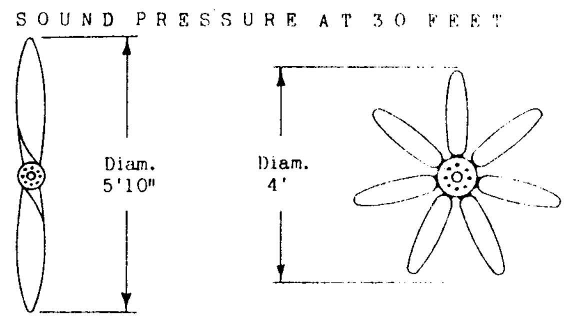


Figure 1.- Test data for two- and seven-blade propeller compared with Gutin theory.

Regier



CUB PROPELLER Sensenich No. 70L45	
I	
R.P.M. --------	2100
Tip Mach No. --	.57
Pitch - - - - -	12.8°
H P - - - - - -	40
Sound Pressure	107 db

NACA HIGH-SPEED BLADES No. 4-308-03	
II	
R.P.M. - - - - - -	2100
Tip Mach No. - - -	.39
Pitch - - - - - -	21.5°
H P - - - - - - -	48
Sound Pressure - -	86 db

Figure 2.- Test conditions for sound recordings.

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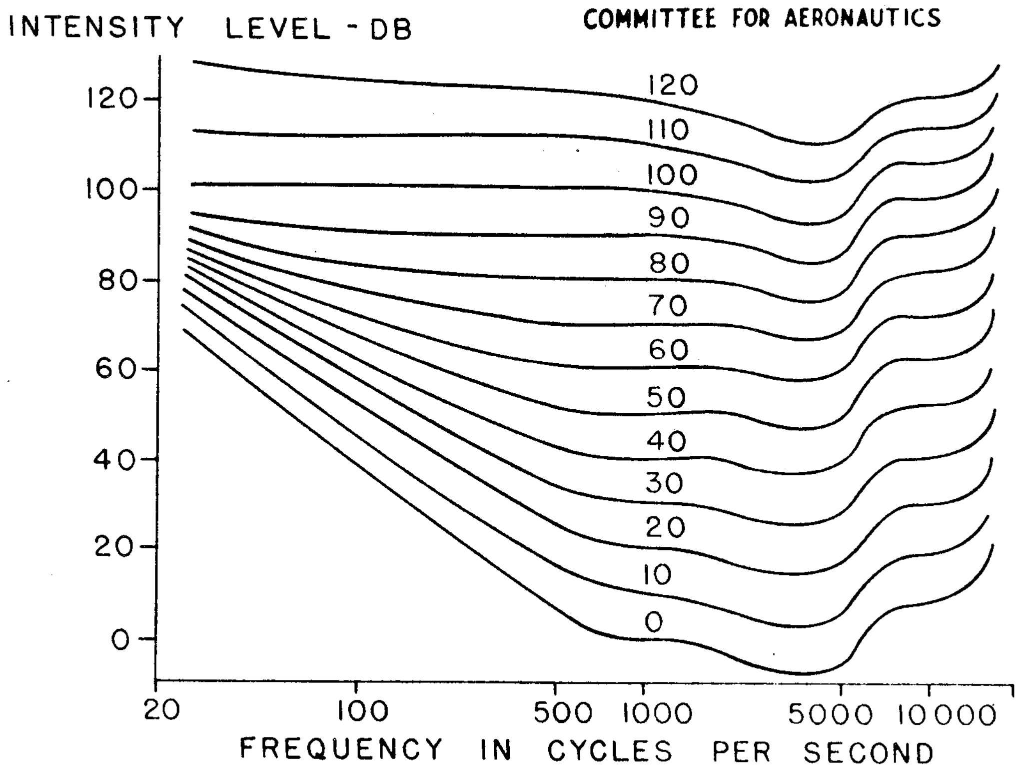

Figure 3.- Loudness-level contours.

PROBLEMS IN THE DESIGN OF PROPELLERS FOR NOISE REDUCTION

By Arthur W. Vogeley

The increased publicity being given to airplane noise indicates that noise reduction is now not merely desirable, but necessary.

It is generally recognized that the worst noise sources are the engine and the propeller. It is relatively easy to quiet the engine by muffling, but it seems that the propeller is much more difficult to handle. It has been shown that propeller noise may be reduced by reducing propeller section speeds and by increasing the number of blades. At the present time, this approach appears to be the only answer to the problem.

Everyone is worrying about what this means to the light airplane. It will mean gear reduction since the only other alternative for reducing propeller section speeds, that of reducing propeller diameter, cannot be tolerated because of the penalty incurred in take-off performance. More blades mean complications, increased maintenance, and the necessity for having a starter on the engine since it will be difficult to crank the engine through safely by hand. It may also require some form of pitch control. The whole business appears to add up to increased cost, weight, and complexity.

Let us, for the moment, however, try to forget the costs involved and just see what happens when we attempt to design a less noisy propeller for a typical personal airplane. The airplane chosen has a top speed of 130 miles per hour, cruises at 120, and takes off at 55. It is powered by a 125-horsepower engine rated at 2550 revolutions per minute. It has a 2-blade direct-drive propeller, 74 inches in diameter. This propeller produces sound at a level of 100 decibels.

The design chosen to reduce the noise is an 8-blade geared propeller. The diameter has been kept constant, because, as mentioned before, reducing the diameter is undesirable from a consideration of take-off performance, and it is assumed that the diameter cannot be increased without radically changing our present airplane configuration.

Let us now see what this 8-blade design means in terms of sound reduction. For this purpose, figure 1 has been prepared from figure 10 of Theodorsen and Regier's report on propeller noise (reference 1). The present 2-blade operating point is located at 100 decibels and at an effective propeller Mach number of 0.6 (the effective propeller Mach number is taken at the 0.8 radius.) On this figure is also shown the new design condition. The effective Mach number has been reduced approximately 50 percent and, according to theory, the sound level should be reduced to something less than 40 decibels. It should be remembered, however, that the present theory neglects vortex noise which becomes a problem when the propeller operates at low rotational speeds. Consequently, the sound reduction to be expected from theory will be optimistic. Nevertheless, it is hoped that by taking such an extreme case the sound level will still remain below the desirable maximum sound level of approximately 60 decibels.

It should not be inferred from this discussion that an 8-blade propeller is the only possible answer to the noise problem. Figure 1 shows that many other solutions are possible. One particular case has just been taken as a first step.

The next two figures will show the relative performance of the 8-blade geared propeller as compared to the original 2-blade direct-drive propeller. Comparisons have been made both as a fixed-pitch and as a controllable propeller. These comparisons are based on calculations using available theory and data. From an aerodynamic standpoint, the requirement of quieter operation introduces no additional problem.

Figure 2 compares the performance of the 8-blade geared propeller with the original 2-blade direct-drive propeller, both operating in fixed pitch. It is seen that the maximum speed and cruising speed remain practically unchanged but the take-off distance with the 8-blade propeller has been increased about 8 percent. Designers may not be willing to accept this reduction in take-off performance, but if it can be tolerated then the necessity for providing pitch control is removed.

Mr. Crigler, in his paper, has shown that with a given propeller diameter, the take-off performance can only be increased by varying propeller pitch. Consequently, it is well to determine how the 8-blade propeller, if made controllable, will compare with conventional controllable propellers.

Figure 3 shows this comparison. Here again, maximum and cruising speeds remain unchanged. By going from a 2-blade direct-drive fixed-pitch propeller to a 2-blade direct-drive controllable propeller, the take-off distance is reduced approximately 22 percent. Changing to an 8-blade controllable propeller, a further reduction in take-off distance of approximately 4 percent is realized.

This brief study has shown that, from an aerodynamic standpoint, a quieter fixed-pitch propeller may be built if some increase in take-off distance can be accepted, but if a controllable propeller can be built then a small improvement in take-off performance can actually be obtained.

As a matter of interest, figure 4 shows approximately what the 8-blade propeller would look like. The blades are more or less conventional, the hub has been made large enough to contain any pitch-change mechanism yet not so large as to increase the engine cooling problem.

I would like now to go back and review some of the problems mentioned earlier and present some random thoughts about these problems.

Gear reduction means increased weight, but, since gear reduction is already necessary to reduce noise, then one can, perhaps, reduce the weight penalty by redesigning the engine to operate at higher revolutions per minute with a resultant increase in horsepower per pound.

At first glance an 8-blade propeller would appear to be four times as heavy as a 2-blade propeller, but, because the rotational speed has been reduced, the centrifugal forces have been reduced, and, because the number of blades has been increased, the thrust and torque loading per blade has been reduced. Consequently, the blade stresses have been reduced enormously and this should make possible the production of considerably lighter blades. A brief stress analysis, considering only thrust, torque, and centrifugal force, indicates that almost any material that will hold its shape would be structurally satisfactory. This may open an entirely new field of blade fabrication.

The design of a pitch-change mechanism for an 8-blade propeller would, at first, also appear to be very complicated. However, if it is remembered that a 2-position propeller is a good substitute for a continuously variable-pitch propeller (at least under the conditions of light airplane operation) then the problem is considerably simplified. Furthermore, because of the low centrifugal forces and low rotational speeds the design for controllability is still further simplified. It appears possible that an extremely simple 2-position mechanism could be built.

Producing a less noisy propeller is a very tough proposition, but it is hoped that this study has shown that if quieter propellers are demanded, one can, by careful consideration of their peculiarities, reduce the penalties that must be paid.

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Vogeley

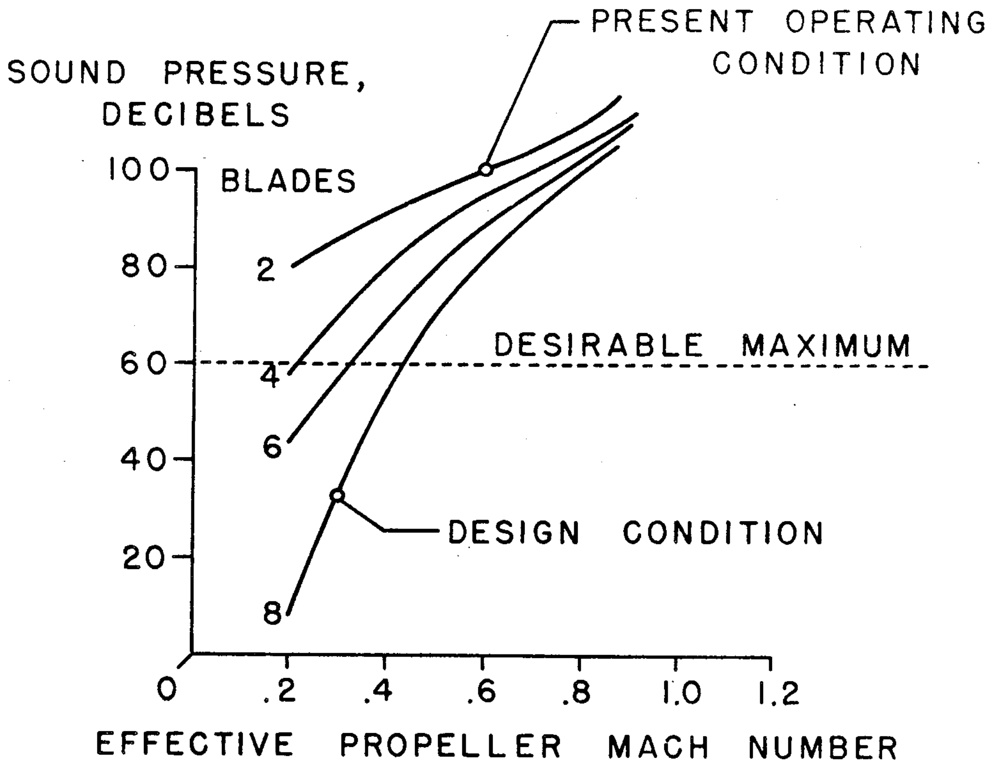

Figure 1.- Sound level as a function of number of propeller blades and Mach numbers.

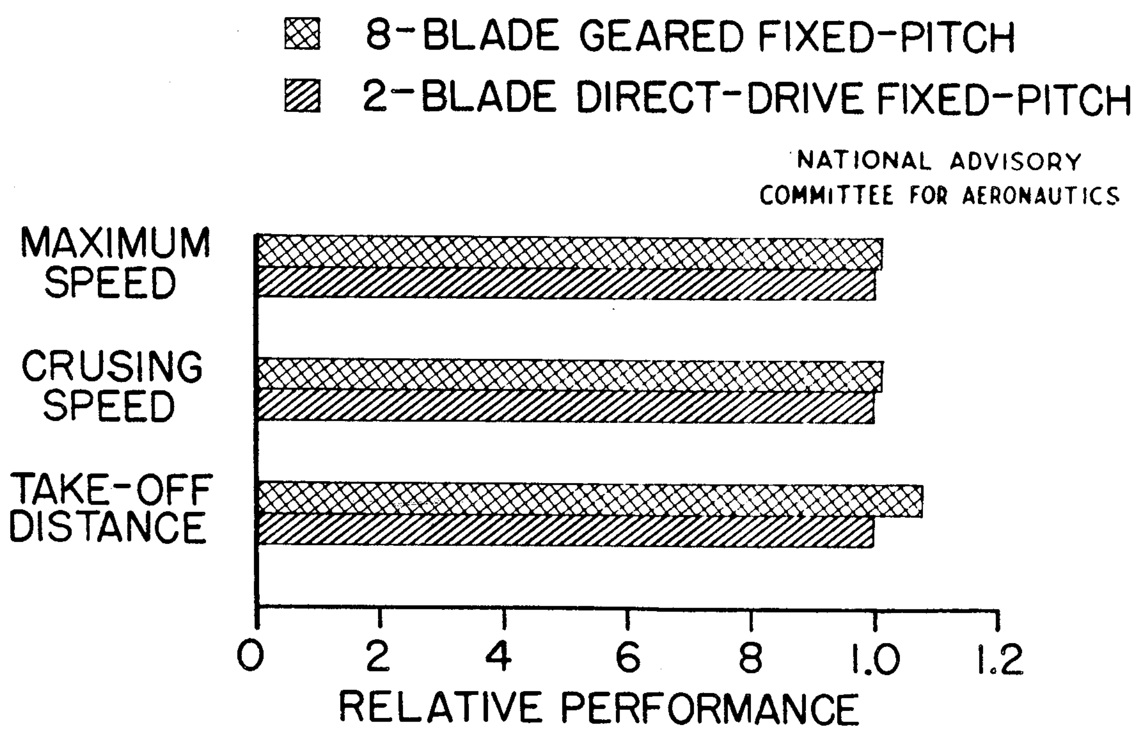

Figure 2.- Relative performance of an airplane equipped with two-blade direct drive and eight-blade geared fixed-pitch propellers.

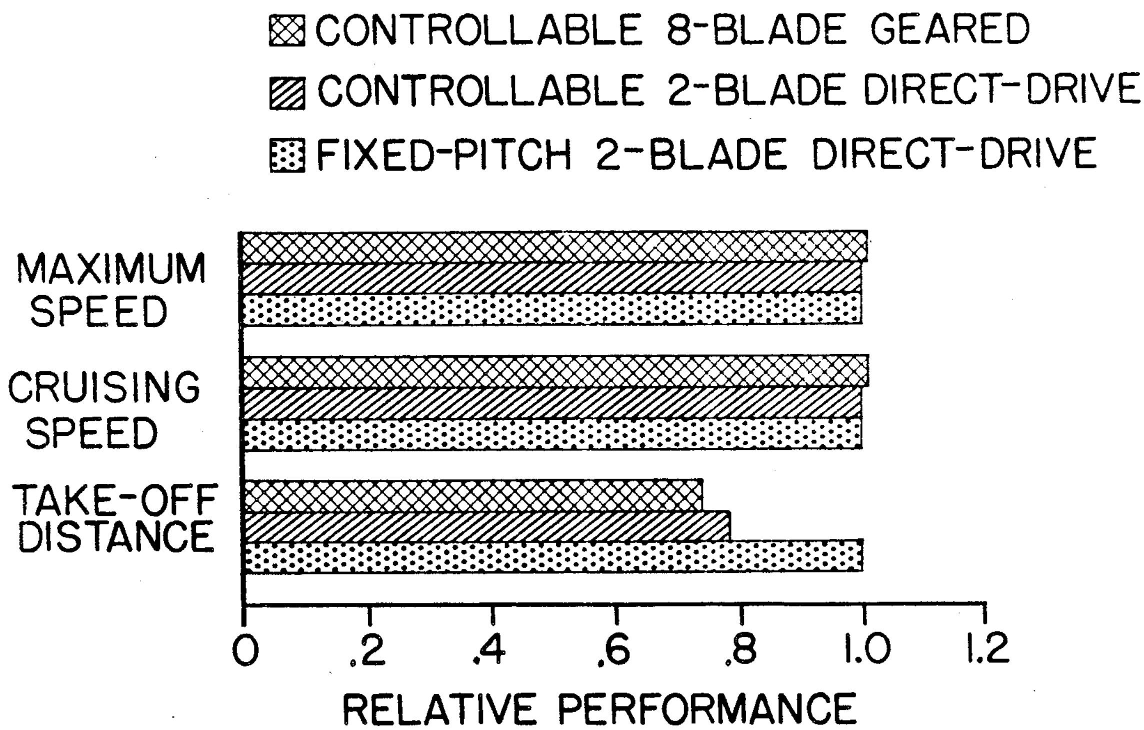

Figure 3.- Relative performance of an airplane equipped with controllable- and fixed-pitch propellers.

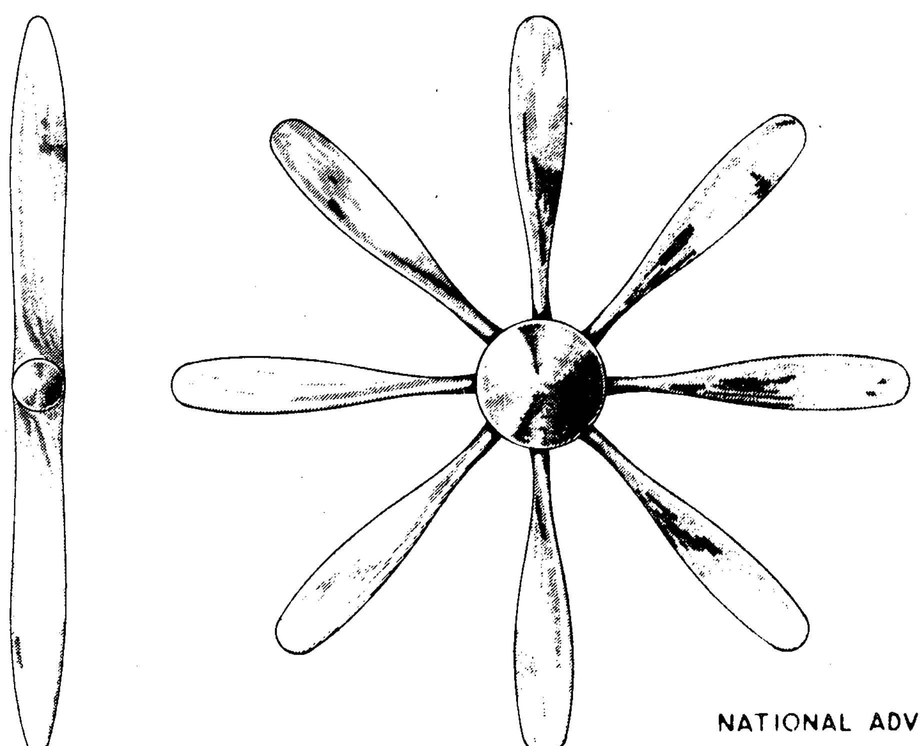

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Figure 4.- Two-blade direct-drive and eight-blade geared propeller configurations.

DRAG CLEAN-UP

PERFORMANCE GAINS BY ATTENTION TO DETAIL DESIGN

By Herbert A. Wilson, Jr.

During the war the most important work of the Langley full-scale-tunnel section was the drag clean-up of military airplanes. In order that this work could be made available for general use, it has been summarized in two reports (references 1 and 2). These reports include the results obtained with 23 airplanes and point out the advantages to be obtained from careful detail design.

Unnecessary drag was found to result from (1) projection of various items outside of a smooth basic contour (2) roughness of surfaces (3) unintentional leakage of air through the airplane structure and (4) the use of large quantities of excess air for various cooling functions.

In order to show the relative importance of these items and the locations in which they are likely to occur and also to acquaint you with the procedure that was used during the investigations, the clean-up of a typical airplane will be described first and later some selected detail cases taken from three airplanes will be given.

The first step of the clean-up is the examination of the airplane. Points at which it is suspected that there is unnecessary drag are studied and analyzed carefully so that changes to reduce the drag can be designed. The airplane is then put in a faired and sealed condition in which all protrusions have been removed or carefully faired, all openings have been closed, and all external leaks have been sealed. Item by item the airplane is returned to its service condition and the drag is evaluated at each step. For example, figure 1 shows such an airplane in the faired and sealed condition. Only the basic combination of streamline body, wing, tail surfaces, and cockpit enclosure remains. We use the drag of this basic condition as a reference and call it 100 percent.

The airplane with all the items added that pertain to the power plant installation is shown in figure 2. Removing the streamline nose that was added on the NACA open-nose cowling and opening the cowling exit so that the engine cooling air could flow added 18.6 percent of the drag of the basic condition. Adding the unfaired carburetor scoop added 3.6 percent, opening the outlet at the back of the accessory compartment added 3 percent, installing the projecting exhaust stacks and opening up the hole through which they projected increased the drag 3.6 percent. Installing the intercooler scoop, and allowing air flow through the ducting system, increased the drag 6.6 percent and installing the oil-cooler system consisting of an external scoop and an unnecessarily large outlet increased the drag 10.2 percent. These items total 45.6 percent of the drag of the basic condition.

Next, the rest of the items necessary to put the airplane in the service condition are added. In figure 3 it is seen that removing seals from the gaps on the cowling flaps increased the drag 5.4 percent, opening the case and link ejector chute gave 1.8 percent, and opening the seal around the edges of the landing-gear doors increased the drag by 1.2 percent. Adding the sand-surfaced slip-proof walkway added 4.2 percent, installing the radio aerial added 4.8 percent, and adding the machine guns and blast tubes added 1.8 percent. This second group of items which include protrusions, roughness, and leakage totals 19.2 percent.

Look at what has happened to the clean airplane we started with! In order to make it useful we have increased its drag nearly 65 percent, mostly by adding items that by themselves do not appear particularly large.

All of this drag, however, is not necessary. Additional tests and careful analysis showed that the drag of the power plant items could be reduced to 26.6 percent and the drag of the roughness and leakage items could be reduced to 2.5 percent, thus saving nearly 36 percent of the drag of the basic condition. On this airplane which is in the 325-mile-per-hour class this amounted to ~~8~~ 26 miles per hour or about an 8 percent increase in speed.

It is particularly important to note that in general these items have drags of only a few percent each. Yet, when taken altogether, they add up to an impressive total. We started with an airplane in figure 1 that was exceptionally clean and in bringing it to a usable configuration unnecessary drag was added along with the drag associated with the necessary functions.

In order to illustrate the principles of clean-up in more detail a few additional items from other airplanes are shown. In all these cases the increments are given in terms of increments of drag coefficient. For your information it is pointed out that a reduction of 0.0002 in the drag coefficient of the 325-mile-per-hour airplane just shown would increase its speed by about 1 mile per hour. On an airplane in the over 400 miles per hour class, the gain from the same drag increment would be 2 miles per hour and on an airplane in the 200-mile-per-hour class, the gain would be slightly less than 1/2 mile per hour.

The upper half of figure 4 shows an engine exhaust stack of the large-bore stovepipe type which was one of two on an airplane. The unfaired protrusion and excess air leakage caused this installation to have high drag. Engine operating tests made both with the original stacks installed and with individual jet type exhaust stacks showed that the combined reduction in drag and increase in thrust of the cleaner installation would increase the airplane speed by approximately 13 miles per hour.

On the lower half of the figure is shown an exhaust-stack installation that does not protrude. Sealing and fairing this exhaust opening decreased the drag by 0.0010. The form drag is not large, but an excessive amount of air was allowed to flow out of this opening at a high angle to the free stream. The drag of this installation could be greatly reduced by reducing the airflow to the minimum that would cool the stacks, and directing the discharged air rearward.

Some makeshift methods of providing engine cooling have been found to cause excessive drag. Opening holes in the cowling either before or behind the engine is especially bad because of the high pressure difference between the inside and outside. In figure 5 an airplane is shown in which holes were cut through the cowling just behind the cylinder baffles in order to remedy unsatisfactory cooling of the engine for the climb condition. The holes failed to provide the desired improvement in cooling and full-scale-tunnel tests showed that the flow disturbances caused by the holes increased the drag coefficient 0.0041 for the cowlings of the two engines of the airplane. The remedy in this case was to close the holes and provide adequate and well designed cooling air outlet area elsewhere.

Also in another case, cutting a hole through the cowling near the nose to get clearance for a machine-gun blast tube increased the drag 0.0029.

Although we ordinarily expect high performance airplanes to have retractable landing gears, figure 6 shows a single case in which a fixed landing gear was used. It was provided with the blunt short-tailed fairing shown by the sketch. Adding the longer fairing shown which extended both nose and tail reduced the drag coefficient 0.0008.

An example of the more normal case with the landing gear retracted is shown in figure 7. The lower half of the figure shows the original production installation for this airplane. Half of the wheel well was left uncovered and the rough hole there caused poor flow and high drag. The cover was then extended to cover the entire well, but was not sealed. This reduced the drag by only 0.0003, but when the 1/8-inch gap around the edge was sealed, the drag was decreased another 0.0009 or a total of 0.0012. This emphasizes the necessity for preventing random leakage of air through the airplane.

Now that you have seen how refinements in the detail design affect high-speed airplanes, let us consider what the application of these principles to personal aircraft means. There are great differences in the characteristics and purpose of military and personal airplanes. The military airplanes flew at speeds of 325 to 450 miles per hour and had wing loadings of from about 25 pounds per square foot up, in contrast to the much lower speed and wing loading of personal aircraft. The increments have been applied only to the top speed, whereas the personal aircraft designer is likely to be interested also in the horsepower required and fuel economy at the cruising speed.

Suppose for example that an airplane could be cleaned up to the extent of reducing its drag coefficient by 15 percent. Based on the results obtained with military airplanes this seems reasonable. Since the speed varies inversely as the cube root of the drag coefficient, this would give a speed increase of approximately 5 percent. Now considering the horsepower required for cruising which varies directly in proportion to the drag coefficient, it is seen that a 15 percent reduction in the cruising drag coefficient would be a 15 percent reduction in cruising horsepower or 22.5 horsepower for an airplane that cruises on 150 horsepower.

In conclusion, I want to point out two things further. First that the gains that will result from careful attention to detail design are likely to be greater than the differences in drag between airplanes of different clean basic configurations. Also, as the speeds and general performance of personal aircraft advance, the importance of good detail design will increase.

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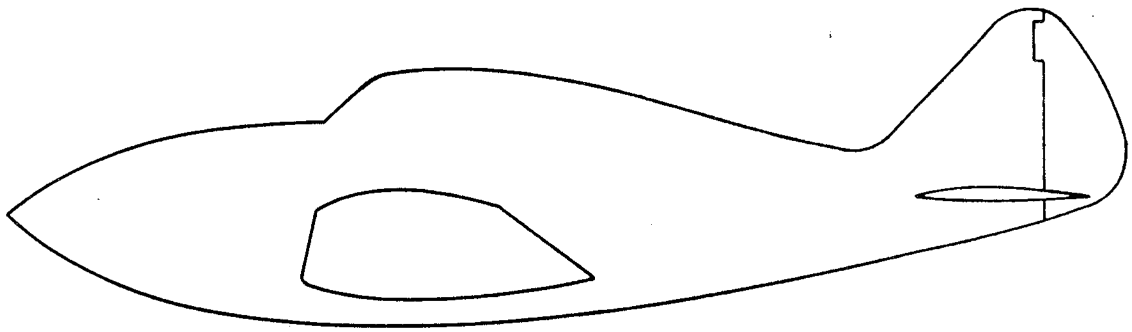

Figure 1.- Clean-up airplane in the faired and sealed condition.

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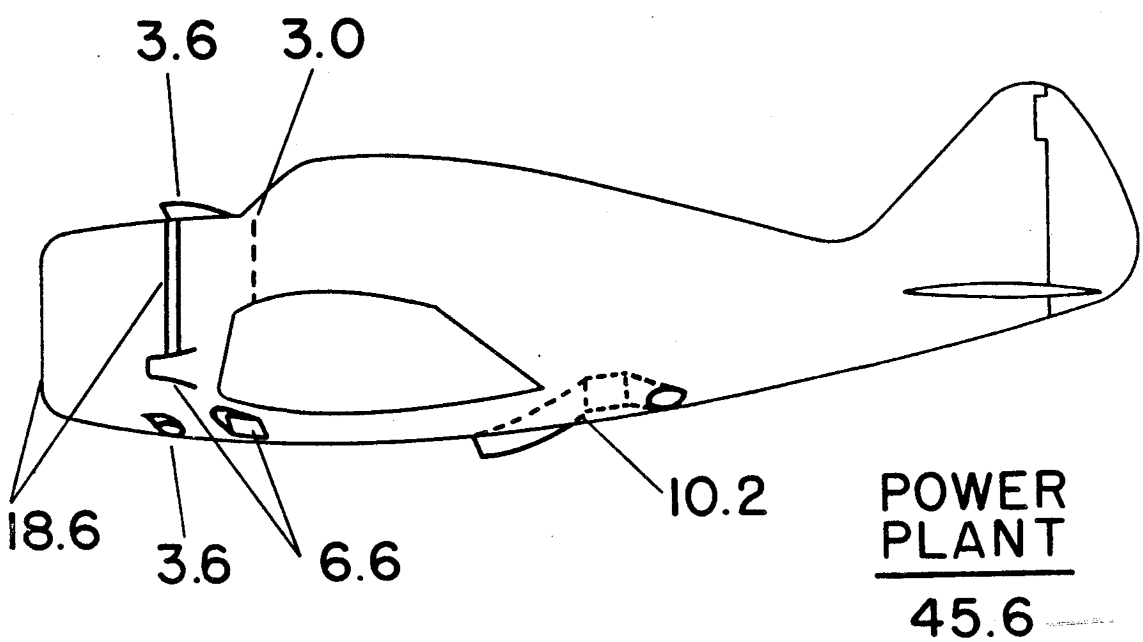

Figure 2.- Clean-up airplane with power-plant-installation drag items added.

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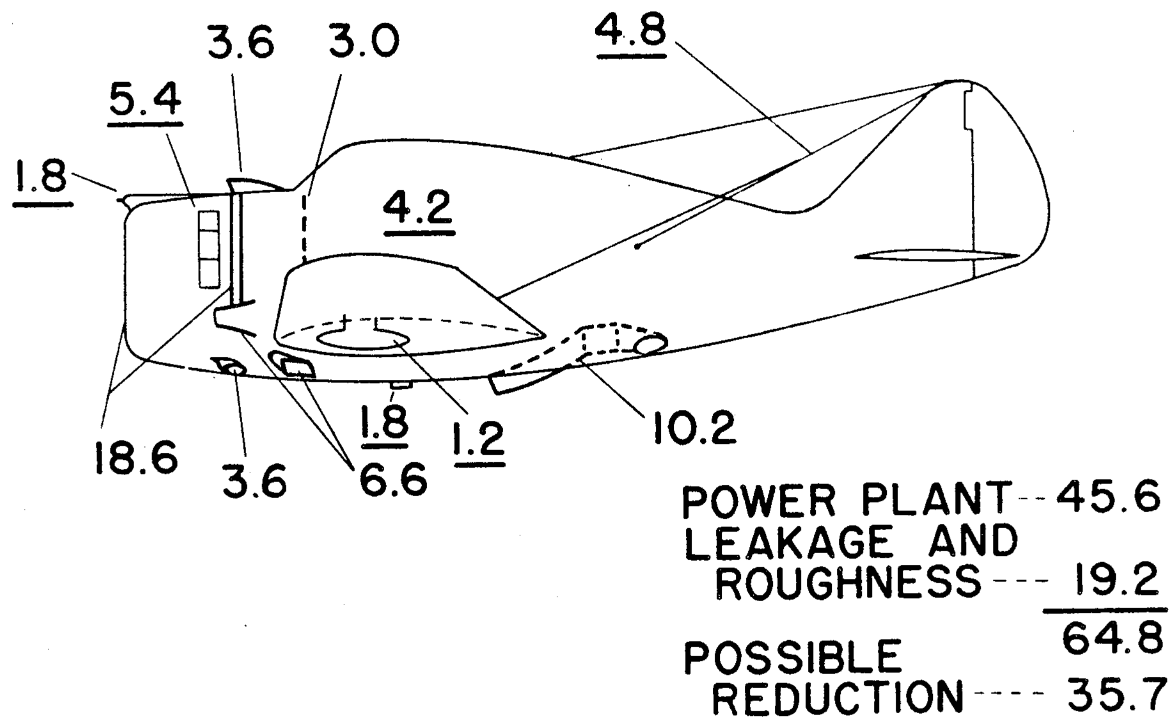

Figure 3.- Clean-up airplane in the service condition.

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Figure 4.- Exhaust-stack installations.

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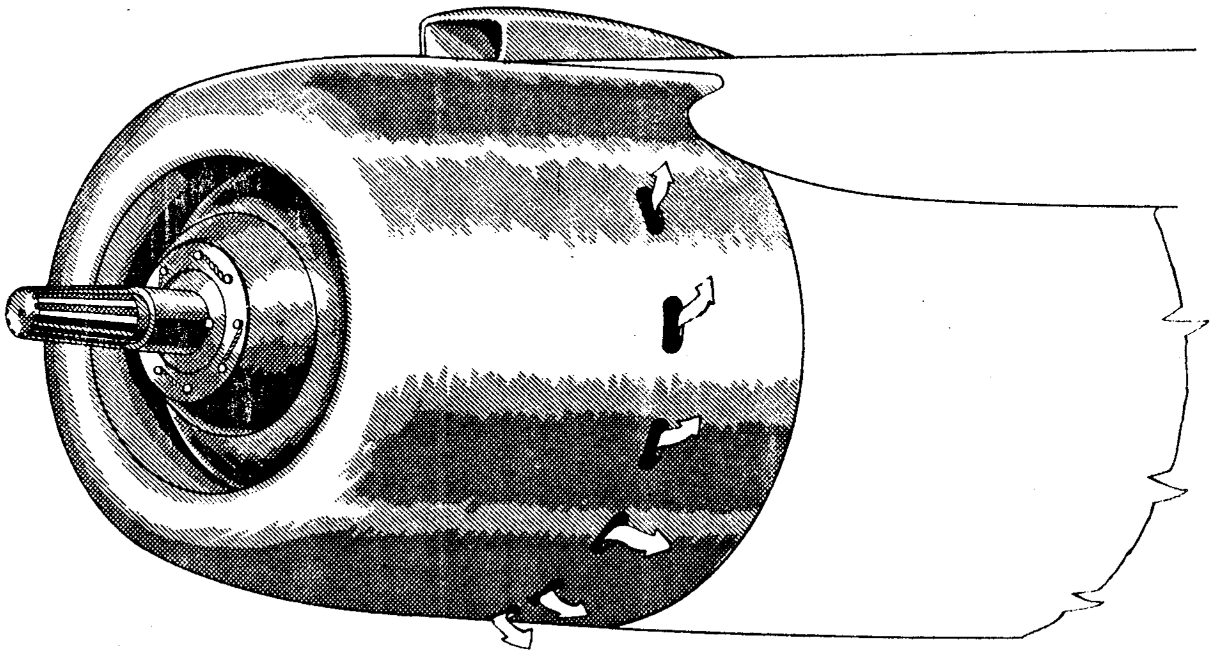

Figure 5.- Cowling with holes cut through behind baffles.

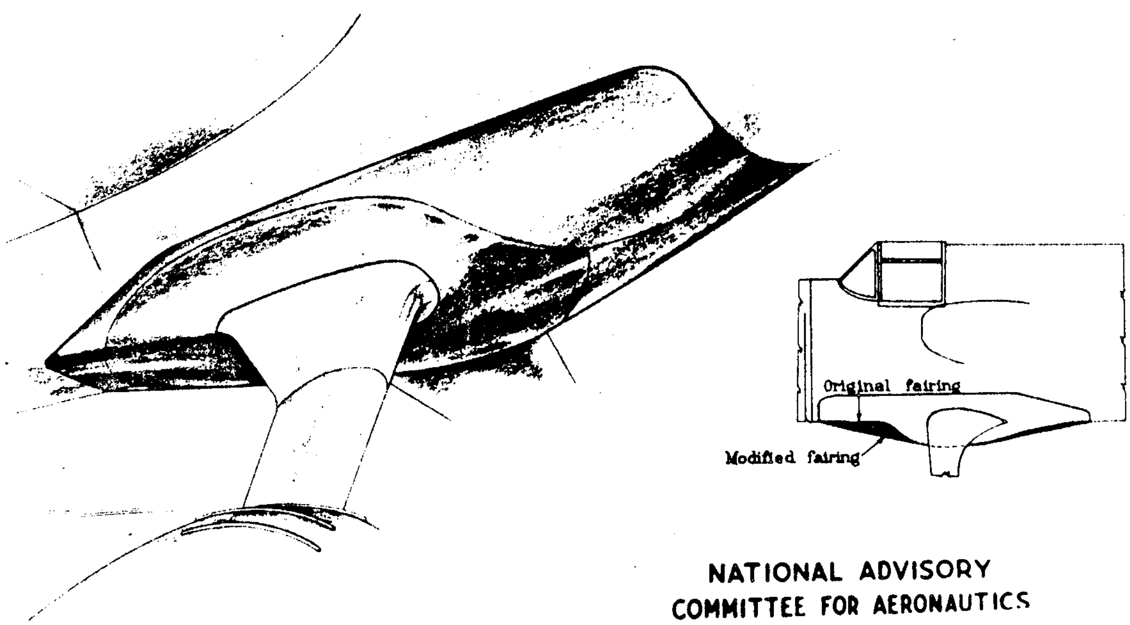

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Figure 6.- Fixed-landing-gear fairing.

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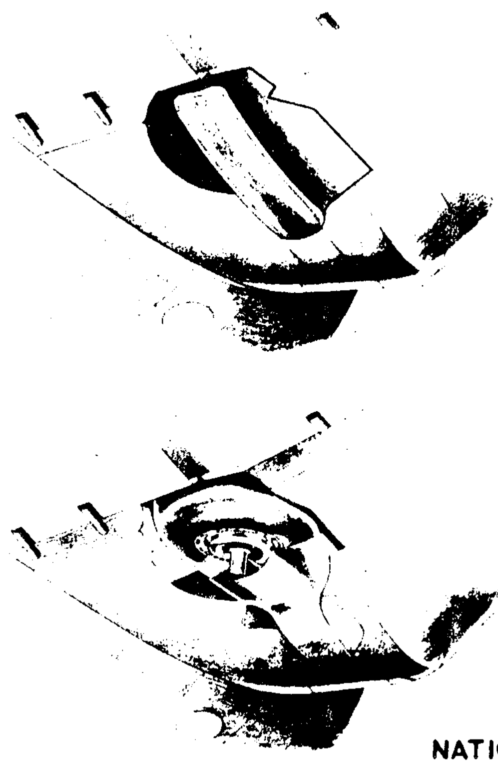

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Figure 7.- Retracted-landing-gear fairing.

AIRCRAFT LOADS

SOME REMARKS ON THE STRUCTURE AND STATISTICS OF ATMOSPHERIC GUSTS IN RELATION TO THE PERSONAL AIRPLANE

By Charles C. Shufflebarger

The flight loads on personal airplanes are governed by the same principles that apply to other types of airplanes. Our research on loads therefore is, in general, applicable to the personal airplane.

For many aircraft, especially those in the nonacrobatic and transport categories, the loads imposed by gusts are critical in the design of the main wing structure. Regardless of the category of the airplane, however, the CAA design requirements on gust loads are the same and the question arises as to whether these requirements are applicable to personal airplanes. While the Laboratory's investigations of gust loads have not been directed primarily toward establishing what those loads are for personal airplanes, the investigations have been, nevertheless, of a fundamental nature, so that much of what has been learned is applicable to light airplanes.

The design of airplanes to withstand gust loads requires not only a knowledge of the intensities of the gusts but it requires further a knowledge of the gust structure - namely the velocity gradients and the spatial dimensions within which the velocity changes take place. Figure 1 summarizes the results of some of our investigations on gust structure. The effective gust intensity U_e is plotted against the gradient distance H. The gradient distance is the distance within which the velocity or intensity of the gust changes from zero to a maximum and it is given in terms of wing chord lengths. The data plotted were obtained from measurements of acceleration and airspeed on three airplanes ranging in size from the small Aeronca light airplane to the XB-15 bomber. Each point on the plot represents the most probable gradient distance of a large number of gusts having the intensity shown, and the plot therefore indicates the size of the gust most likely associated with any given value of the gust intensity.

It can be seen that the data all fall into the same pattern within the limits of the scatter regardless of the size of the airplane. It may thus be concluded that the gust structure on which the current design requirements are based is as applicable to the small airplane as to the larger transport types.

Figure 2 further substantiates the applicability of our fundamental gust investigations to the small airplanes. Statistical determinations of the frequency distribution of gust intensity are often of value and shown here are the relative frequency distributions of the effective gust velocity. These frequency distributions are simply a plot of the relative frequency of occurrence of gusts whose intensities lie within certain limits. For example, a relative frequency of 10^{-3} means that one out of 1000 gusts encountered (on the average) will have an intensity equal to or greater than the corresponding gust intensity shown.

The two frequency distributions shown were taken from reference 1 and represent the frequency distributions as determined by measurements with a Lockheed XC-35 airplane and by measurements with an Aeronca C-2 airplane. The measurements with the XC-35 were taken within thunderstorms at all altitudes between about 5000 and 35,000 feet. The measurements with the C-2 on the other hand were taken in turbulence caused by shearing of the wind close to the ground. The frequency distributions, as can be seen, are nearly identical notwithstanding the considerable differences between the airplanes on which the measurements were made and between the meteorological conditions associated with the turbulent regions.

The actual choice of design values of gust intensity depends on the operating conditions, which determine the turbulent regions of the atmosphere through which the various classes of airplanes will fly. This question involves the statistics of operating practices, and our statistical studies have centered largely about the problem of the transport airplane. There is some evidence to show, however, that the design gust intensity of 30 feet per second now incorporated in the design requirements is reasonable as applied to the personal airplane category. This evidence is shown in figure 3. The absolute frequency or the number of gusts of a given intensity that will be encountered within a given period of operation depends upon the ratio of the miles flown in rough air to the total operating miles flown. The curve in figure 3 taken from reference 2 shows the result of absolute frequency determinations for transport airplanes. The absolute frequency is represented by the inverse quantity distance M_o in terms of operating miles required to encounter a gust of a given intensity. The curve shown is given in terms of unit chord length and it applies specifically to average transport conditions for which the path ratio R or ratio of miles flown in rough air to total operating miles is 0.10. The distance required to encounter one gust of a given intensity for any value of the chord length and for any value of the path ratio can be determined from the curve and by application of the formula shown on figure 3.

The operating miles M and the operating times M/V shown in the inset table were determined in the following manner: An arbitrary obsolescence operating time for a typical transport airplane was selected. The value chosen was 20,000 hours and the chord length and operating speed of the airplane were taken as 12 feet and 200 miles per hour, respectively. The path ratio was, of course, taken as 0.10 corresponding to average transport operating conditions. These numbers, when incorporated in the formula and with M_o taken at a value corresponding to a gust intensity of 30 feet per second in effect stipulate that this value of the gust intensity which corresponds to the limit load factor will be equalled or exceeded a given number of times within the selected operating life. Now, if we choose a personal airplane and permit it to encounter the same number of gusts equal to or exceeding 30 feet per second within its operating life, we can solve for the operating life and compare it with that originally chosen for the transport airplane. The path ratio R, however, requires some modification from the value applicable to average transport conditions. We suppose that the personal airplane will, on the whole, be operated at somewhat lower altitudes than is common to average transport operations. Some of the earlier transport data taken when operating altitudes were in the neighborhood of 3000 to 5000 feet indicate that for such conditions a path ratio of 0.20 is more nearly applicable than 0.10 to the personal airplane operations. When this value of path ratio is used in computing the operating time for the personal airplane, the value of about 8900 hours of operations is found, something less than half of the operating life chosen for the transport machine.

The only significance we attach to this result is that the operating lives of the personal and transport airplanes work out in a proper relative order. If we had found, for example, that the operating life for the personal airplane worked out to be substantially larger than that of the transport airplane rather than substantially smaller, we would be in a position to say that the design gust intensity of 30 feet per second was probably too large to be correct for the personal-airplane category. In this connection it might be noted by reference to the flight-path curve, that small or moderate changes in the gust-intensity result in fairly large changes in the operating flight path. For example, a change in gust intensity of only about 6 percent will result in a change of the flight path by a factor of 2.

In conclusion, considering gust structure, the relative frequency distribution of gusts, and the operating times for the transport and personal airplane, the design gust velocity of 30 feet per second appears to be of the right order for personal as well as for transport airplanes. It should also be mentioned that the gust load will probably be the design loads for a number of airplanes, especially for the nonacrobatic types.

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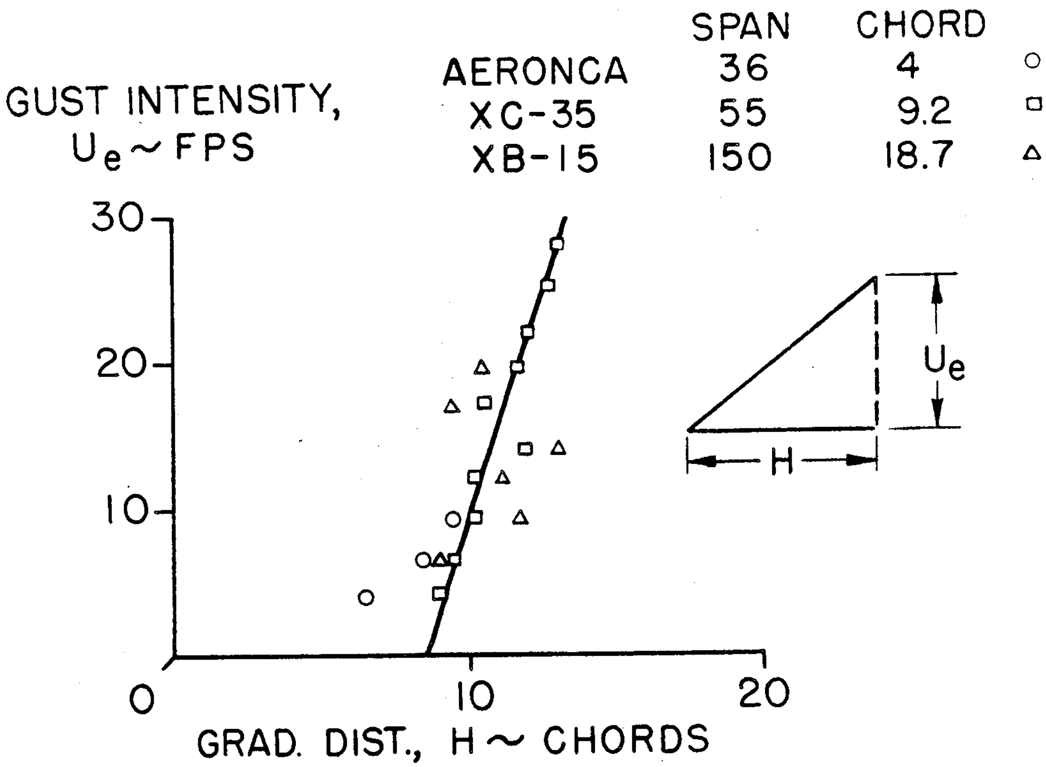

Figure 1.- Gradient distance of gusts determined from measurements with three airplanes.

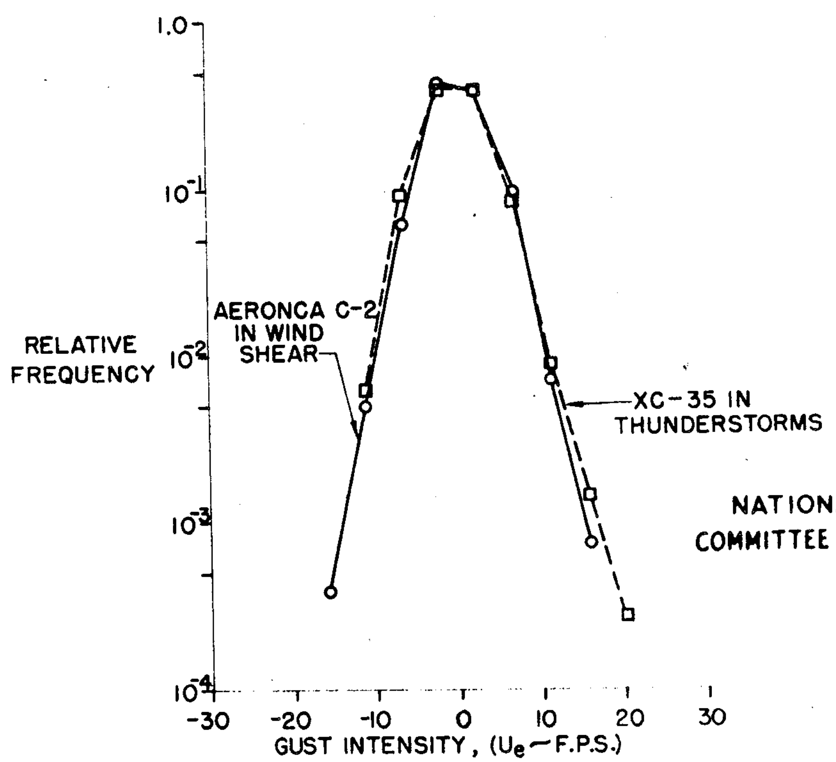

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Figure 2.- Relative-frequency distribution for samples selected at random from turbulence surveys with Lockheed XC-35 and Aeronca C-2 airplanes.

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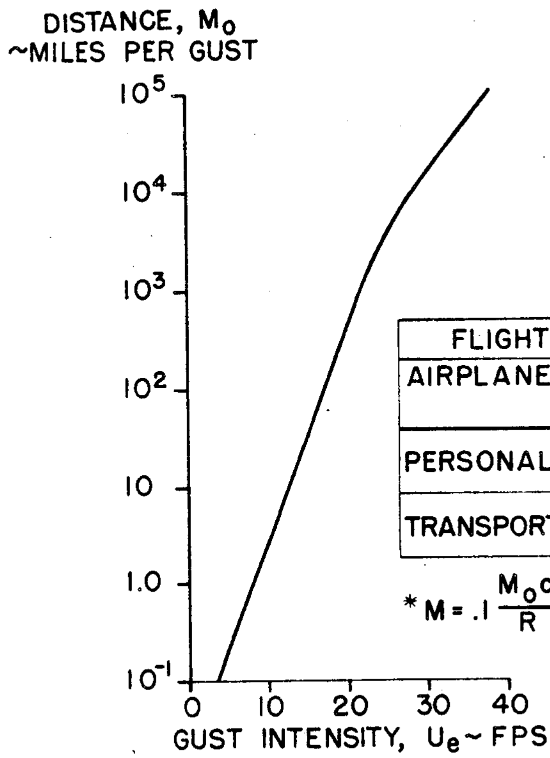

FLIGHT TIMES FOR $U_e \geq 30$ FPS					
AIRPLANE	c FT	V MPH	R	M* MI.	M/V HR.
PERSONAL	8	150	0.2	1.3×10^6	8930
TRANSPORT	12	200	0.1	4×10^6	20,000

$* M = .1 \frac{M_o c}{R}$ FOR ONE GUST $\geq U_e$

Figure 3.- Miles per gust equal to or greater than selected values for airplanes with mean geometric chord of 1 foot. Average airline operating conditions.

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MANEUVERING LOADS ON VERTICAL AND HORIZONTAL TAIL SURFACES

By Henry A. Pearson

During the war a number of load investigations were undertaken whose results are applicable to the personal airplane. These investigations had to do both with the determination of the load magnitudes and the load distribution. Owing to limitations of time we will confine our remarks to what has been done and what is available for the determination of the loads on horizontal and vertical tail surfaces for airplanes in the acrobatic category.

With regard to the determination of the magnitude of the horizontal tail load, much has been accomplished to rationalize the problem and to relate the tail loads to the geometry and stability characteristics of the machine. In this connection, a convenient method has been developed (reference 1) by which it is possible to compute the horizontal-tail load and the wing load for any given elevator deflection. The method requires a knowledge of certain stability derivatives which may either be obtained from the available wind-tunnel results or may be computed by ordinary engineering procedures.

Figure 1 shows the agreement which is obtained between the calculated and experimental tail loads for the XP-51 airplane in a pull-out made at about 315 miles per hour. The calculated results are based on the actual elevator motion used in flight. It will be seen that the agreement of the loads on both the wing and tail is exceptionally good.

It might be mentioned that in this case, the necessary stability derivatives were obtained from model tests. Other comparisons (see reference 2) which have been made for airplanes of various sizes also indicate good agreement so that with regard to the horizontal tail load, it may be stated that a method exists which is not only rational but which gives good agreement under usual conditions. In fact, such a method has already been incorporated into the design requirements for military machines.

Although a suitable method exists, a fundamental part in applying it for computing the design load on the horizontal tail has been in the specification of the control motion to be used. Some work has been done in this field and the rates of control movement and the forces which a pilot is capable of exerting on the stick have been investigated. (See reference 3.) The results of these investigations have made possible the specification of conservative stick motions which are within the physical capabilities of the pilot and which can be made to tie in with the specified load factor. For design, the most conservative control motion is the one in which the pilot pulls the stick back as rapidly and as far as is possible, following which it is moved forward rapidly in time to check the maneuver at the specified load factor.

Some effort has also been directed toward rationalizing the vertical tail load problem and two pertinent reports have been issued. (See references 4 and 5.) As in the case of the horizontal tail, the methods which have been developed depend on both the geometric and stability derivatives of the airplane. Figure 2 shows how well the calculated and experimental vertical tail loads agree for a rudder kick made at 250 miles per hour with the P-40K airplane. The dotted line shows the computed load using the experimental rudder motion. It is seen that as in the case of the horizontal tail, a reasonable agreement was obtained between theory and experiment. Therefore, it seems evident, as was the case with the horizontal tail, that providing the rudder motion and aerodynamic characteristics are known, the correct tail load can be calculated.

In order to illustrate what happens to the vertical tail load with different types of rudder motion, the results of some computations are presented in figure 3. While the illustrative examples apply to a large flying boat flying at 200 miles per hour, the results are qualitatively applicable to any type of airplane.

In the upper left hand corner of figure 3, we have an instantaneous rudder kick of 1°. Directly below we see that the load on the vertical tail at first increases by 1200 pounds and then, as the airplane swings toward the equilibrium position, the load reaches a maximum value of 1500 pounds, finally after several oscillations, attaining a steady value of about 700 pounds. Thus, for this particular machine the ultimate design load for which the tail was designed would be reached by an instantaneous deflection of about 19° at the initial condition, of only 15° at the midpoint, and of 33° at the endpoint.

It has been thought that in the fishtailing maneuver any vertical tail surface could be broken providing the maneuver were continued sufficiently long. The top right hand portion of figure 3 shows a rudder operation of 1° made in resonance.

with the airplane's natural period. The lower right hand curve shows the tail load computed for the above rudder oscillation. It is seen that in this case the vertical tail load, after about two cycles, reaches approximately 2700 pounds per degree of deflection, whereas, if carried on indefinitely, it would ultimately reach 2800 pounds. With this particular maneuver, only about 9° rudder movement would cause the tail load to reach the ultimate design value. It can also be seen that the tail load approaches a limit and does not increase indefinitely as might be supposed. By comparing the two cases, it is seen that the type of rudder motion has much to do with the vertical tail load. In fact, it is possible to build up the tail load to larger values than are shown by imposing combinations of the two types of rudder motion.

The logical specification of a rudder motion, however, is more difficult than the specification of an elevator motion because there is no well established lateral acceleration to which the rudder motion can be linked and also because resonant effects are likely to be more pronounced on the vertical tail load than in the case of the horizontal tail load. Large vertical tail loads also arise from other maneuvers such as a rolling pull-out(see reference 5) as well as from direct rudder action. Since this subject is covered in one of our recent reports (see reference 5) and since rudder motion is not a main factor in rolling pull-outs, the subject will not be covered at this time.

Summarizing then, we can say that with regard to tail loads adequate methods are available for computing the loads which are imposed for a given control motion. With regard to control motions, we have reached the point where the specification of a reasonable elevator motion is possible. Although we have not reached a similar point for the rudder motion, statistical data of the types of motions used by pilots are being obtained.

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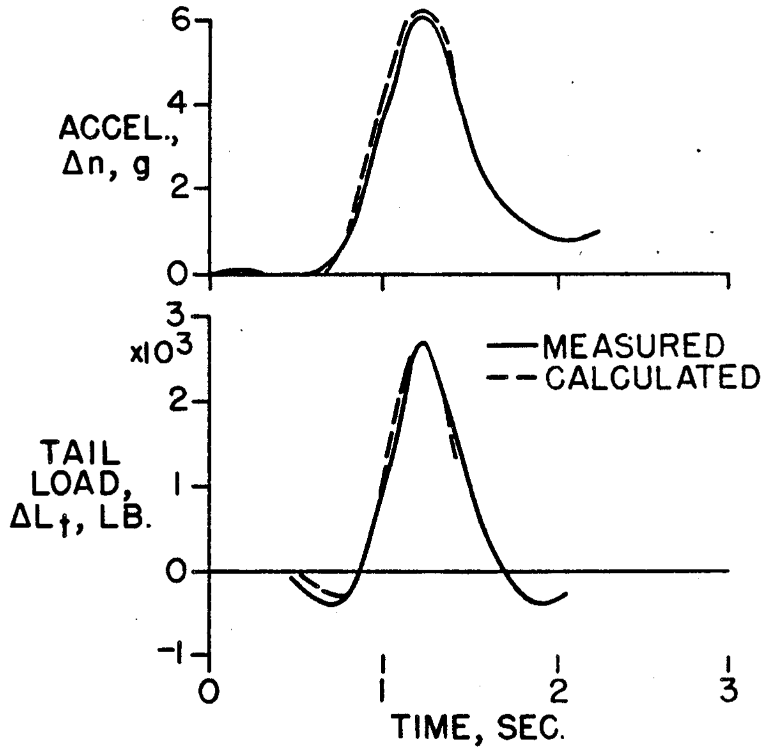


Figure 1.- Comparison of calculated and measured tail loads and accelerations for an XP-51 airplane in a dive pull-out.

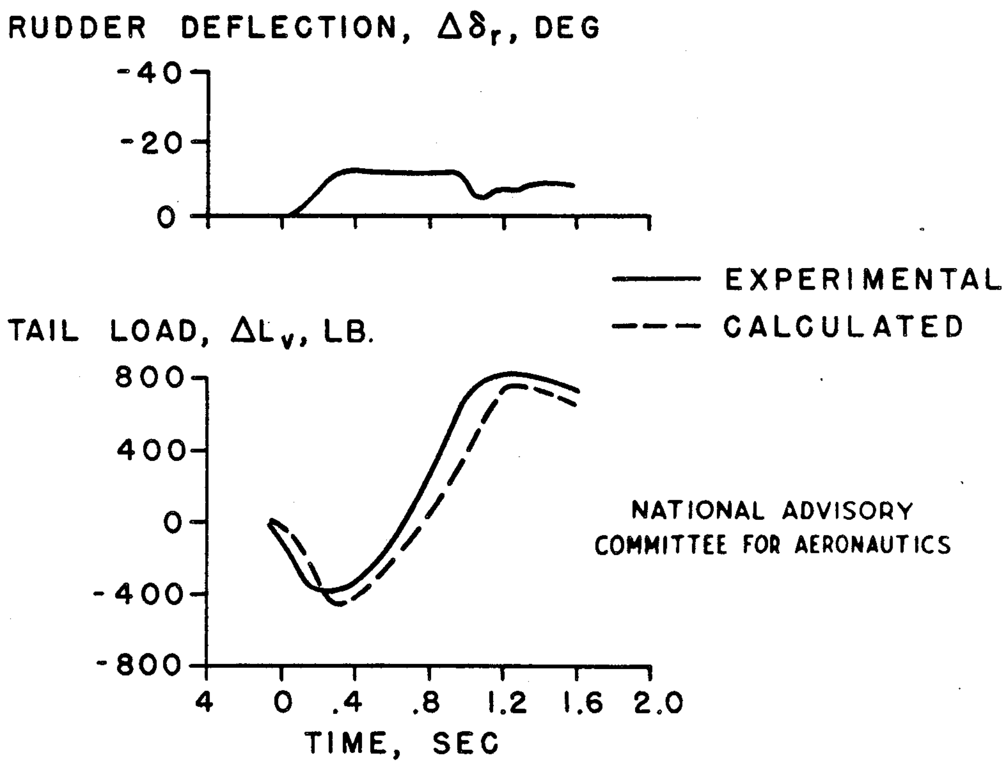
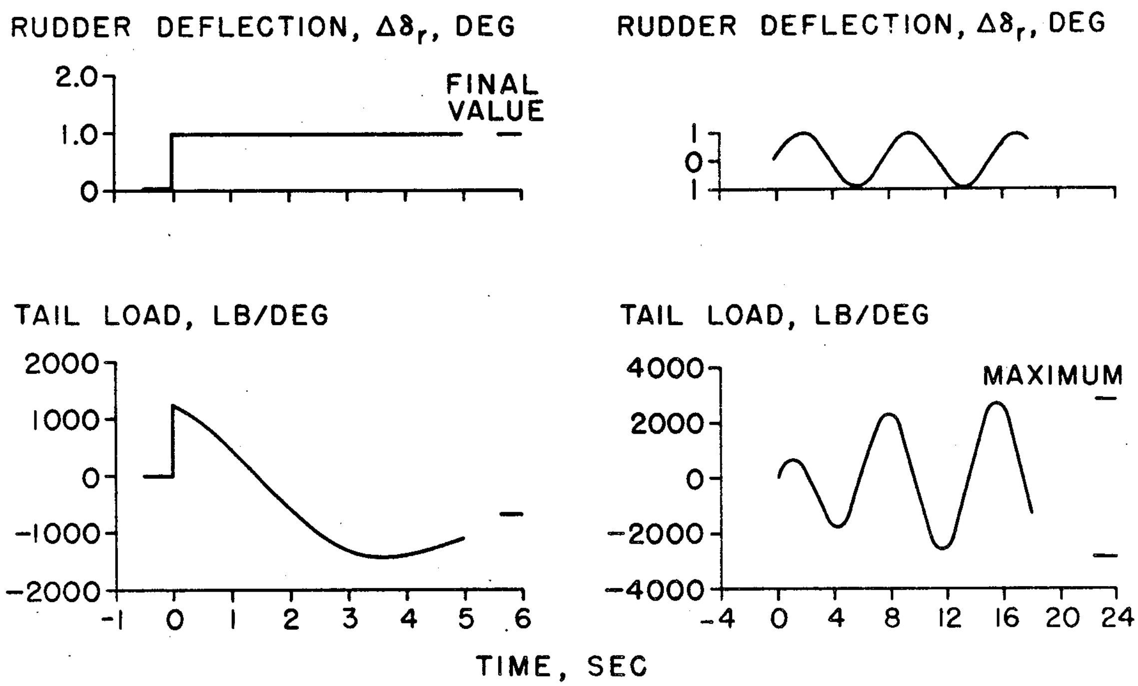


Figure 2.- Comparison of calculated and measured vertical-tail load on a P-40K airplane due to rudder motion.

Pearson



(a) RUDDER KICK

(b) FISHTAIL

Figure 3.- Calculated vertical tail loads for a flying boat in rudder kick and fishtail maneuvers.

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SEAPLANES

SEAPLANE AND AMPHIBIAN DESIGN

By John B. Parkinson

INTRODUCTION

During the war, research on seaplanes in the Langley tanks was mainly concerned with the large multiengine flying boats used for patrol and transport missions. Attention was centered on hydrodynamic stability and spray problems associated with the overloading of existing types, and on the design of very large long-range transports like the Martin Mars and the Hughes Hercules (reference 1). Little direct experience was gained with the smaller seaplanes and amphibians; in fact, it appeared that hydrodynamic problems related to their design and operation were minor or were being adequately solved on the basis of existing information.

Basically, however, the designer of water-based airplanes is confronted with the same general requirements regardless of size or configuration - the attainment of adequate take-off performance, hydrodynamic stability and controllability, and seaworthiness with the minimum of penalties in aerodynamic performance. Different combinations of wing loading, power loading, and arrangement alter the relative importance of the various hydrodynamic requirements, but they must all be properly considered to produce the most successful design. In this light, the data and conclusions in the Wartime Reports from the tanks are applicable to the personal-airplane field, and provide additional background for post-war developments.

CONVENTIONAL HULLS

Hydrodynamic Resistance

Relative importance.- With low power loadings, water resistance is a relatively unimportant criterion in hull design as compared to stability and spray. With high power loadings, about 15 or 16 pounds per horsepower, it tends to limit the useful load, prolong the take-off run, and thus accentuate other hydrodynamic problems.

For the same wing and power loadings, small seaplanes tend to have poorer take-off performance than large seaplanes because they operate at higher Froude numbers; that is, higher speeds in relation to their dimensions. The relatively low wing loadings usually associated with personal-owner airplanes, on the other hand, have a favorable effect by reducing the load on the water and the take-off speeds.

Effect of length-beam ratio.- The water resistance at the hump speed and at high planing speeds is largely affected in opposite directions by the hull size for a given gross weight. Consequently, the effects of the size of the hull must be eliminated as far as possible when investigating the effects of length-beam ratio alone on take-off performance.

One method of holding size constant when comparing the resistance at various length-beam ratios is to hold the product of the length and beam constant. This procedure is analogous to holding the wing area constant when comparing the drag at various aspect ratios.

A typical effect of increasing length-beam ratio while holding the length-beam product constant on the total resistance during take-off is shown in figure 1. (Data is taken from reference 2.) Increasing the length-beam ratio from 5.2 to 7.8 by increasing length and decreasing beam as indicated decreased the resistance at both the hump and high planing speeds, with a resulting decrease in take-off time of 12 percent. With a higher power loading and smaller take-off thrust, the improvement in take-off performance would of course be correspondingly greater.

Aerodynamic Drag

Significance.- The aerodynamic performance of seaplanes is usually inferior to that of the corresponding landplanes because the bulk of the seaplane is greater to provide spray clearance for propellers and aerodynamic surfaces. Improvements in performance must be directed, not only toward reduction in the drag coefficients of the hulls or floats, but also toward reduction in size by the use of more hydrodynamically efficient bottoms.

Effect of length-beam ratio.- The improvement in take-off performance with increase in length-beam ratio offers the alternate possibility of reducing drag by reducing hull size without undue impairment of the hydrodynamic qualities. Research along these lines has been conducted by a wind-tunnel and tank investigation (unpublished) of a related series of hulls with length-beam ratios of from 6 to 12, designed to have comparable hydrodynamic characteristics but smaller size as the length-beam ratio is increased.

The resulting hull forms are shown in figure 2 along with the pertinent aerodynamic characteristics as obtained from tests in the Langley 300 MPH 7-by 10-foot tunnel. The coefficients are all based on the wing area of the same hypothetical seaplane and represent the absolute changes that would be obtained by interchanging the hulls on the assumed design. Increasing the length-beam ratio from the conventional 6 to the novel 12, while maintaining comparable hydrodynamic performance, reduces the drag by some 20 percent. The slopes of the pitching- and yawing-moment curves (C_{m_α} and C_{n_ψ}) indicate a possible reduction in horizontal-tail area, but a corresponding increase in vertical-tail area, when the length-beam ratio is increased. The over-all trend on the aerodynamic performance, however, appears to be favorable. The hydrodynamic data for the series are not complete but indicate the length-beam ratio 12 hull to be as good as or better than the length-beam ratio 6 hull.

Hydrodynamic Longitudinal Stability

Dynamic models.- Research on the porpoising of seaplanes has been carried out in the tanks with powered dynamically similar models. These models provide a close simulation of actual seaplane operation, and their characteristics can be directly correlated with pilot's experience (reference 3). They demonstrate clearly to the design engineer the relative importance of various hydrodynamic criteria, and the integrated effects of design parameters on these criteria. A number of reports on investigations of dynamic models are available (references 4, 5, 6, and 7), which cover most of the important hull variables.

Take-off stability.- All conventional seaplane hulls are unstable on take-off if trimmed below the lower-porpoising limit or above the upper-porpoising limit. For a given hull, the trim depends on the position of the center of gravity and the elevator moment. These characteristics result in a range of stable positions of the center of gravity for take off analogous to the aerodynamic operating range of center-of-gravity positions. The stable center-of-gravity range can be obtained in a straightforward manner in the tank or in flight tests, and thus becomes a useful criterion for hydrodynamic stability.

Typical data establishing the hydrodynamic center-of-gravity range are shown in figure 3 in which the maximum amplitudes of porpoising encountered during accelerated take-offs are plotted against the fore-and-aft position of the center of gravity for one elevator position. As the center of gravity is moved forward past the point where lower-limit porpoising is encountered, the amplitude increases progressively to a dangerous value; likewise, if the center of gravity is moved aft past the point where upper-limit porpoising is encountered, the amplitude also builds up progressively. Assuming a maximum allowable amplitude of 2°, the hydrodynamic stable range at this elevator setting is from 30 percent to 37 percent of the mean aerodynamic chord. This stable range may be shifted forward or aft by upward or downward movement of the elevator, respectively. Extreme use of the elevator to extend the stable range, however, is not favored by pilots.

Because of the large effect of the fore-and-aft position of the step on the hydrodynamic moments about the center of gravity, the hydrodynamic center-of-gravity range is a useful criterion for this important design parameter. If the hydrodynamic range does not correspond to the aerodynamic operating range, it may be shifted fore and aft by moving the step. In practice, the step is located largely on the basis of the desired forward hydrodynamic limit. Unless all the aerodynamic and hydrodynamic moments can be closely estimated, the proper location of the step is best determined by dynamic model tests in the tank, in which all the moment-producing components of the airplane are closely simulated.

Landing stability.- Several flying boats and amphibians have had a serious form of hydrodynamic instability during landing (and in some instances during take-off) called "skipping." The skipping motions in trim and vertical position, particularly for small seaplanes, are rapid and not under control of the pilot, and the seaplane is often thrown clear of the water below flying speed (reference 8).

Research with dynamic models in the tank has shown that skipping is primarily a function of the landing trim and speed, and that the cause of the instability is insufficient ventilation of the afterbody area directly behind the step. Consequently, sufficient ventilation in this area, either by an adequately deep step or by auxiliary ventilation ducts, must be provided to attain stable landing characteristics (reference 9).

Figure 4 from reference 9 illustrates the conclusions stated. The data were obtained from a dynamic model having insufficient depth of step for adequate landing stability. At low landing trims where the main step touched first, the landings were smooth with little amplitude in trim. At higher trims where the afterbody touched first, the model skipped violently at all trims up to the stall. An increase in depth of step from 5.5 to 7.2 percent of the beam eliminated the skipping for all practical purposes. The corresponding data for the amplitudes of vertical motion show similar trends with landing trim and depth of step.

The relation between depth of step and landing stability provides a useful criterion for the minimum depth of step required. Similarly, the relative merit of other variations in step design influencing the effective depth, or the hydrodynamic effects of fixed step fairings to reduce drag, can be evaluated in terms of an important, easily recognized hydrodynamic quality. The chief hydrodynamic penalty for a full step fairing is the same as for a shallow step; that is, stable landings can be made only at low trims and high landing speeds (reference 10).

Spray Characteristics

Relation of hydrodynamic loading and forebody dimensions.- It has been common experience in the tanks that the length of the forebody has a pronounced effect on the cleanness of running at taxying speeds, also that changes in beam alone have a relatively minor effect, even though the changes represent large variations in beam loading. This experience and a correlation of the known spray characteristics of multiengine flying boats (reference 11) have led to an empirical relationship between the gross-load coefficient and the forebody-length-beam ratio as follows:

$$C_{\Delta_0} = \frac{\Delta_0}{wb^3} = k\left(\frac{L_f}{b}\right)^2 \qquad (1)$$

where

Δ_0 the gross load, pounds

w density of sea water, 64 pounds per cubic foot

b beam, feet

L_f length of forebody from bow to step, feet

and k is a coefficient that varies approximately linearly with the severity of the spray thrown at low water speeds over the bow, into the propellers, and against the flaps. Low values of k, 0.05 to 0.07, usually lead to satisfactory low-speed spray characteristics over a wide range of hull proportions; while high values, 0.09 to 0.11, usually correspond to undesirable spray which limits the over-all usefulness of the seaplane. There have been exceptions to the rule, but, in general, the spray coefficient k has proven to be a useful criterion for the selection of forebody proportions and dimensions of a multiengine configuration.

It is interesting to note that from equation (1):

$$k = \frac{\Delta_0}{wL_f^2 b}$$

Hence k and the severity of the spray remain more or less the same for different length-beam ratios, if the ratio Δ_0/L_f^2b is kept the same. A constant value of the product L^2b where L is the length from bow to sternpost, also results in constant trim tracks and resistance over wide variations in length-beam ratio. The validity of the products L^2b and L_f^2b as fundamental hydrodynamic parameters has been further established from tank tests of the series of hulls referred to previously.

Effects of powered propellers.- Observations of the undesirable spray of heavily loaded flying boats and their tank models show clearly the necessity of powered propellers for tank tests to adequately reproduce the full-size spray patterns and to determine fully the effectiveness of spray-control devices. For normal configurations, the rotating propeller blades and slipstream greatly increase the height and volume of the spray at taxying speeds, and reduce the amount striking the tail surface at higher speeds (reference 12).

An investigation of the effects of various parameters on the spray in propellers of a flying boat has shown that the application of power reduces the minimum gross load at which spray strikes the outboard propellers by some 20 percent. The minimum gross load increases approximately linearly with upward movement of the propeller position (reference 13).

Typical data establishing the speed range as a function of load over which spray was observed in the propellers of a twin-engine dynamic model are shown in figure 5. In general, there are two regions corresponding to light spray which does little damage and main spray which is of most concern. Both sets of boundaries converge with decrease in gross load and come together at the loads for which the propellers are clear. Similar data may be obtained for spray striking the flaps and tail surfaces.

The speed range and minimum load for spray in the propellers or for spray striking other components of the airplane, while not direct measures of the severity, are useful criteria for the effectiveness of spray-control devices. In addition, they are easily determined either in the powered-model tests or full size, and thus provide comparable data for correlation purposes.

Spray control.- Various means of keeping spray out of the propellers and off the flaps have been evaluated in the tanks and have been applied full size. In general, forebody spray strips which have transverse discontinuities to break up the spray instead of smoothly faired transverse sections to build up a pressure peak at the chine are best. One of the most effective forms of spray strips investigated was a vertical plate extending downward from the forebody chines. Increasing the trim by increasing the sternpost angle or the forebody-afterbody length ratio also has a favorable effect on the spray.

The best method of insuring satisfactory spray and seaworthiness for a seaplane is to maintain the proper relationships between the weight to be carried and the dimensions of the hull. A sound and not over-optimistic estimate of the maximum gross weight in the preliminary design stage is essential to the optimum design of the hull lines (reference 1). The degree of conservatism in this respect depends of course on the type of seaplane. The most efficient aerodynamic design can probably be obtained by use of a small overloaded hull combined with spray-control devices as required to obtain tolerable spray characteristics.

THE NACA PLANING-TAIL HULL

The results of the type of research on conventional hulls described have lead to the development in the tanks of an unconventional form called the planing-tail hull, which may be of interest in the personal-airplane field. This hull has a deep pointed step faired in plan form and a long afterbody extending back to the tail surfaces. Tests of models have indicated that the planing-tail hull has much lower resistance than a conventional hull (references 14, 15, and 16); hence it is useful for high power loadings or for reduction of hull volume. In addition, unpublished results of investigations with powered dynamic models have indicated it to have superior dynamic stability on take-offs and landings and wind-tunnel tests have shown its drag to be of the same order as for conventional hulls.

A profile of the new hull form and a comparison of its resistance with that of a conventional hull are shown in figure 6. In the comparison, the planing-tail hull has a load-resistance ratio at the hump speed of 6.5, the highest that has been recorded, as compared to the usual value of 4.6 for the conventional hull. At speeds near take-off, the resistance of the planing-tail hull, by virtue of the large afterbody clearance, is of the order of 50 percent of that of the conventional hull.

CONCLUSION

The hydrodynamic resistance, stability, and spray characteristics of a new design may be estimated approximately from the generalized data and experience found in the NACA reports. Further improvements in the personal-airplane field may be achieved by more attention to the form and size of hull, balancing of the longitudinal moments, depth and location of the step, and relations of the loads and proportions.

SYMBOLS

C_{Δ_0}	gross-load coefficient $\left(\frac{\Delta_0}{wb^3}\right)$
Δ_0	gross load, pounds
w	weight of sea water, pounds per cubic foot
b	beam of hull, feet
C_R	resistance coefficient $\left(\frac{R}{wb^3}\right)$
R	water resistance, pounds
C_V	speed coefficient $\left(V/\sqrt{gb}\right)$
V	water speed, feet per second

g	acceleration of gravity, feet per second^2
L/b	length-beam ratio
C_{m_α}	variation of pitching-moment coefficient with angle of attack
C_{n_ψ}	variation of yawing-moment coefficient with angle of yaw
C_m	pitching-moment coefficient, foot-pounds $\left(\frac{M}{qcS}\right)$
M	pitching moment, foot-pounds
q	dynamic pressure, pounds per square foot
S	wing area, square feet
c	mean aerodynamic chord of wing, feet
C_n	yawing-moment coefficient $\left(\frac{N}{qb_w S}\right)$
N	yawing moment, foot-pounds
b_w	wing span, feet
$C_{D_{min}}$	minimum drag coefficient $\left(\frac{D}{qS}\right)$
D	drag, pounds

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Parkinson

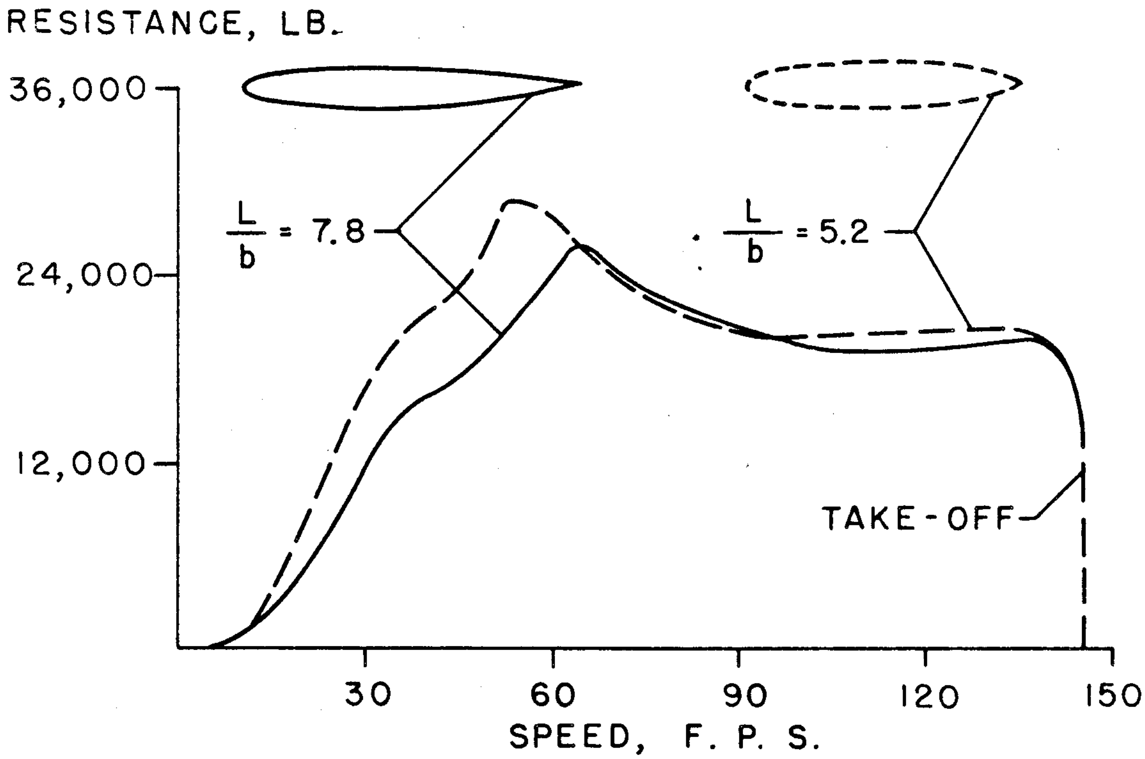

Figure 1.- Typical effect of increasing length-beam ratio with constant length-beam product on take-off resistance.

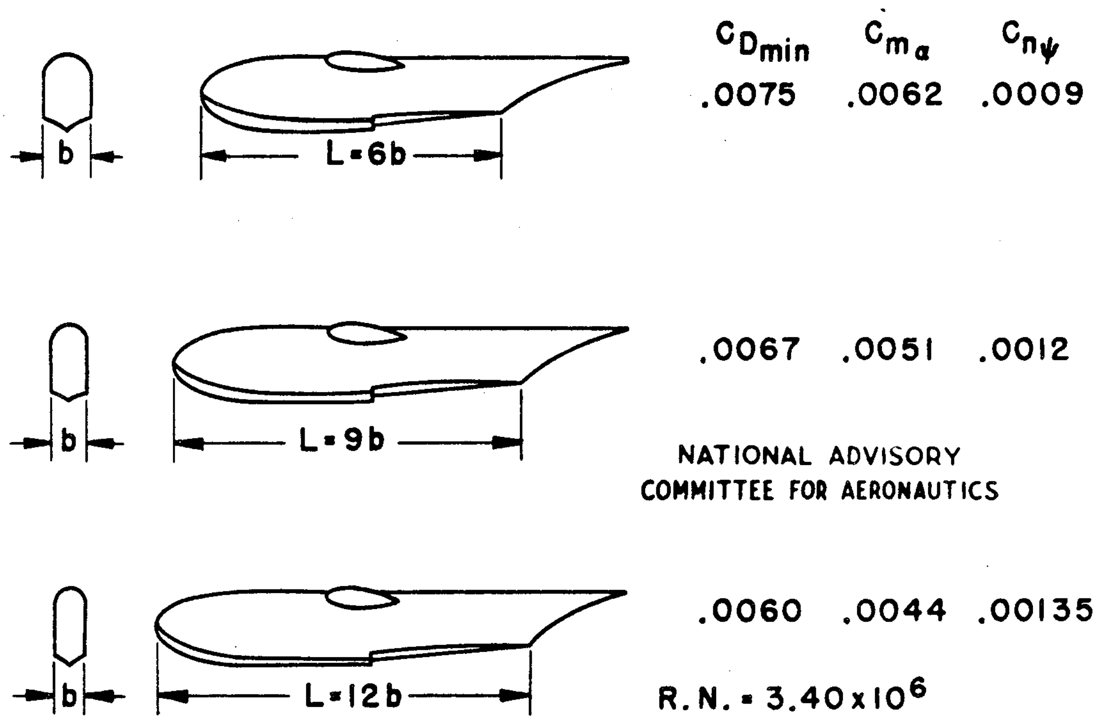

Figure 2.- Effect of increasing length-beam ratio on the aerodynamic characteristics of hulls as determined in the Langley 7- by 10-foot 300-mph wind tunnel.

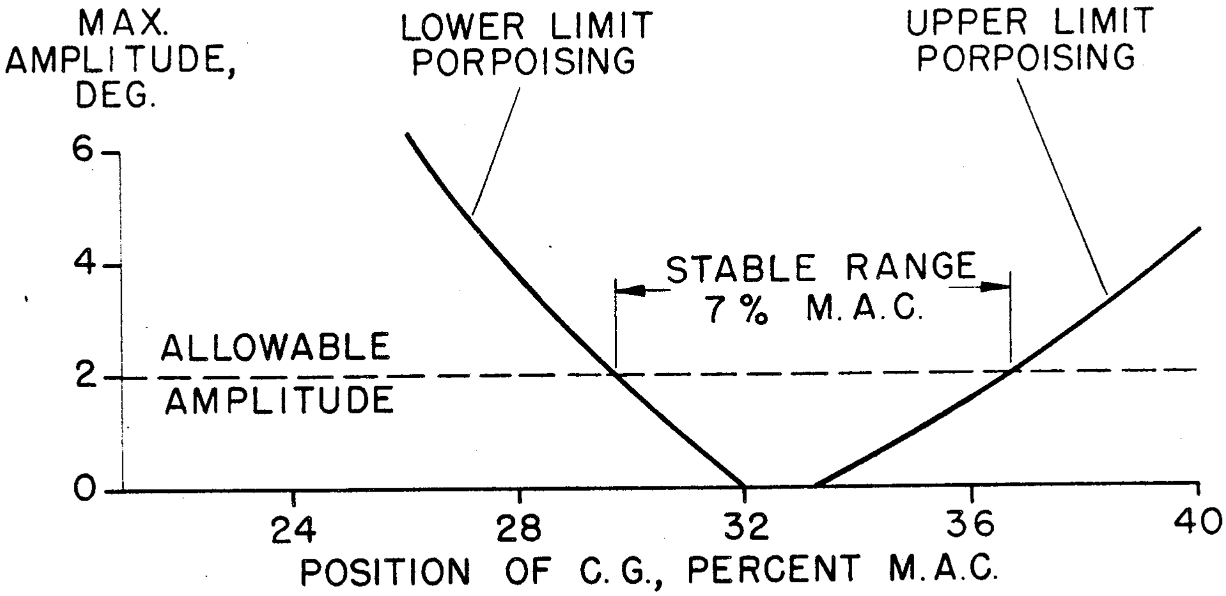

Figure 3.- Typical effect of longitudinal position of the center of gravity on amplitude of porpoising during take-off with constant elevator setting.

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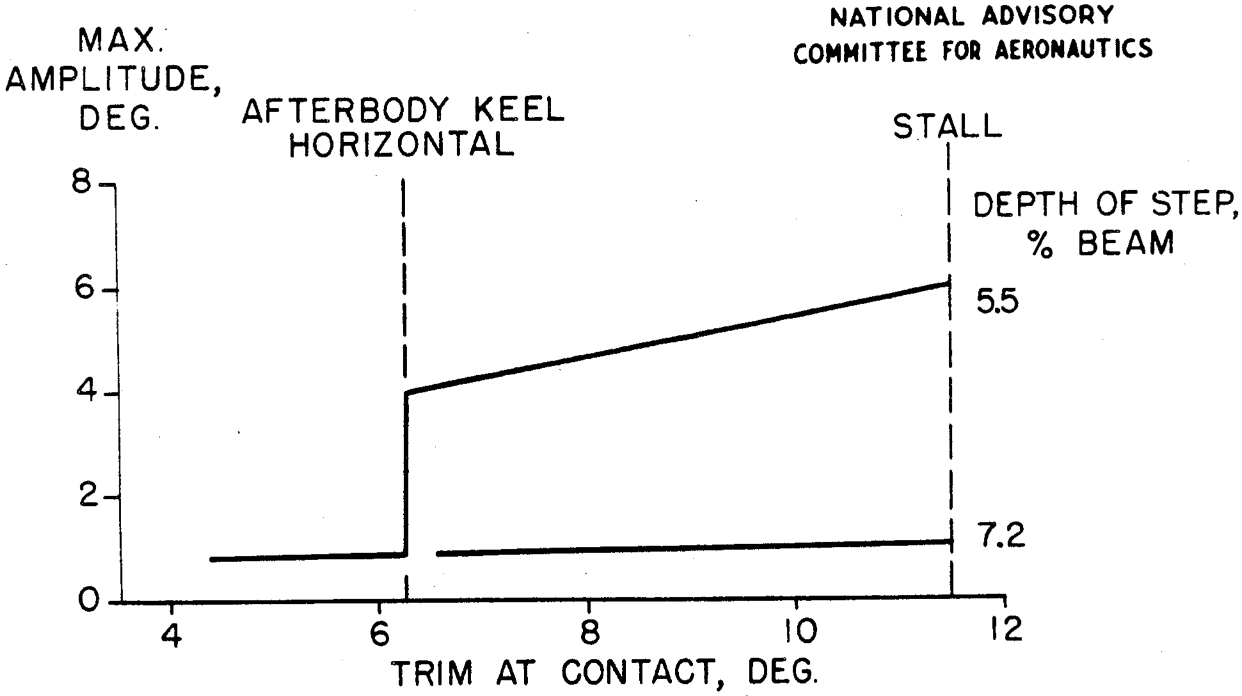

Figure 4.- Typical effect of contact trim and depth of step on amplitude of skipping during landing.

Parkinson

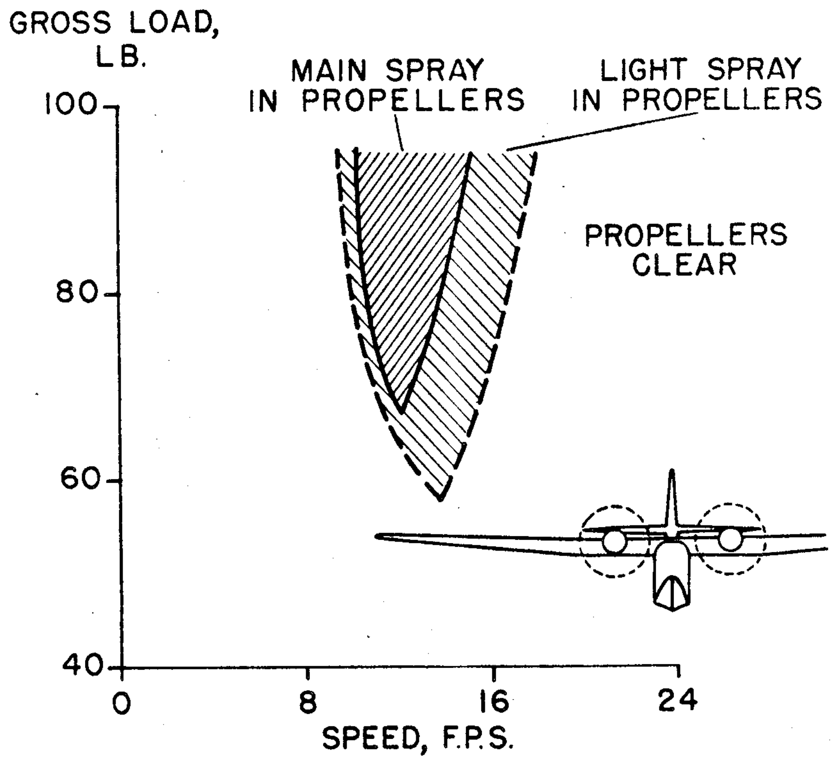

Figure 5.- Typical effect of gross weight on the range of speeds for spray in the propellers of a model of a twin-engine flying boat.

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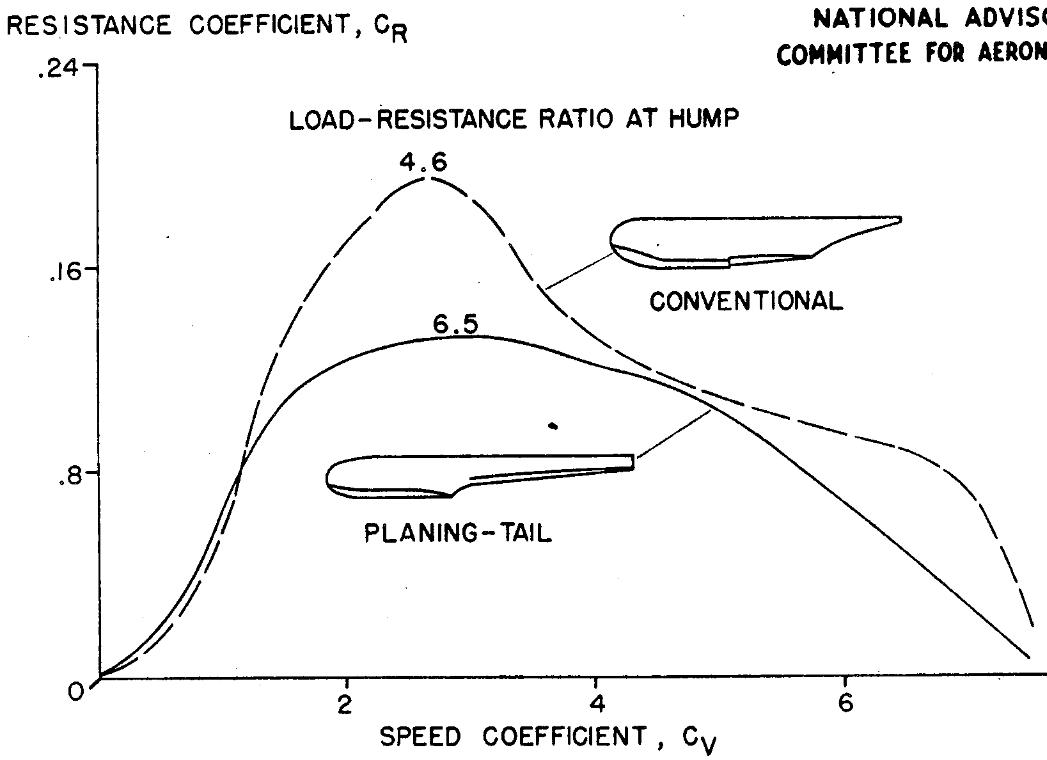

Figure 6.- Typical comparison of resistance at best trim of conventional and planing-tail hulls. Gross load coefficient $C_{\Delta_o} = 1.0$.

POWER PLANTS

AERODYNAMICS OF POWER-PLANT INSTALLATIONS

By Kennedy F. Rubert

The radial form of air-cooled engine has had such extensive military application that the NACA radial engine cowling has been widely used and has undergone intensive development. Other engine configurations have found favor in personal aircraft, but the use of optimum cowlings for these has been less general.

The fundamental principles of cowling design and methods of analysis are the same for all engine configurations, and the experience acquired in developing cowlings for military aircraft may be applied to the improvement of cowlings for personal aircraft. The application of these principles will be illustrated by a discussion of a research reported in reference 1, which dealt with the development of cowling for an airplane of the light airplane class, powered with a six-cylinder flat air-cooled engine. Similar investigations for representative V-type air-cooled engines are reported in references 2 and 3 and problems of liquid-cooled installations are treated in references 4 and 5.

The investigation chosen to illustrate these principles was the development of a power-plant installation of a six-cylinder Franklin air-cooled flat engine of 130 horsepower for a radio-controlled target airplane. This project was conducted in the fall of 1941 for the Army Air Forces, Materiel Command, by the Fleetwings Company and the NACA working in close cooperation. A Fleetwings model 33 airplane flown at Langley Field, Va. by company personnel was used as a flying test bed for the cowlings and baffles designed and constructed under the direction of the NACA.

The airplane before modification is shown in figure 1. Cooling air was supplied to the top of the engine through nose inlets on each side of the propeller shaft. The cooling air passed downward across the unbaffled cylinders and was discharged at the sides and bottom of the cowling. This installation did not provide sufficient cooling capacity to meet the more stringent requirements of the proposed target airplane.

The revised installation, designed to increase the available cooling pressure differential and to decrease the air-flow and pressure-drop requirements, is shown in figure 2. A single inlet for cooling air was provided directly below the propeller axis. Total pressure in excess of that of the free stream was obtained by locating the inlet as far as possible from the axis of rotation, benefiting thereby from the more active part of the propeller disc. The inlet opening was made as large as possible commensurate with maintenance of stable intake flow in the high-speed level-flight condition. In this way, the greatest possible degree of conversion of dynamic pressure to static pressure was taken ahead of the inlet, at high efficiency, thereby greatly relieving the demand for internal diffusion which cannot be done as efficiently. The inlet location chosen is favorable to stability of flow at low-inlet velocity ratios, due to the absence of boundary layers and hub wakes, and an inlet velocity ratio of 0.3 was found practical. In less favorable circumstances, as where spinner wakes must be ingested by the inlet, higher inlet velocity ratios are necessary for stable flow.

Immediately aft of the inlet proper a diffuser is provided which, although quite short because of space limitation, effectively still further reduced the velocity of the incoming air, and introduced it into a generous pressure space, or plenum chamber located directly under the engine. The velocity of flow in this chamber was so low that the motivating force for discharge of air was almost entirely static pressure, and the air could be distributed in all directions with great uniformity. Application of this principle is essential in the design of systems requiring distribution of flow among a multiplicity of passages, if a complicated and inefficient system of guide vanes is to be avoided.

Air from the plenum chamber was forced upward across the cylinders, in reverse of the direction of the original installation. Baffles were fitted to the cylinders to restrict the flow to that usefully employed in passing over the cooling fins, and to eliminate all avoidable pressure loss.

The air discharged from the baffles flowed to a fixed exit through a relatively small collecting passage on the top of the engine. The favorable pressure gradient which occurs in accelerated flow is a powerful deterrent to flow losses, and consequently, the discharge chamber could be much smaller than the intake space. The exit shown was placed at the point where the greatest depression of static pressure of the external air stream could be obtained in the climb condition. This location would be objectionable except in a pilotless craft because of the danger of fouling the windshield, and for piloted aircraft side or bottom outlets would be necessary, although less effective.

For the limited number of flight conditions for which high performance was desired in the target plane the fixed exit opening could be employed, but for most applications controllable exits are far preferable. A hinged flap performs two functions, the first of which is to adjust the area of the exit opening such that the minimum needed cooling-air flow is obtained. The use of a fixed opening, designed for adequacy in climb results in gross overcooling in high-speed level flight with consequent excess cooling drag. For example, reference 6 shows that an exit to give adequate cooling for a 100-mile-per-hour sea-level climb is responsible for a cooling-drag power in level flight at 220 miles per hour of seven times that necessary. The second function of the hinged flap is to create a local depression of the external static pressure as it is extended, increasing thereby the pressure available for cooling. For deflections up to 10° or 15° to the surface approaching the limit of flap effectiveness, this can be accomplished without serious increase in drag. Excessive extensions, however, may so increase the external drag that the loss of flight speed decreases the available dynamic pressure more than the decrease in static pressure obtained at the exit. In such a case, extending the flaps slows the airplane down and overheats the engine. Reference 7 reports tests of a representative flap installation which caused external drag equal in amount to all of the rest of the drag of the airplane, at extensions which are quite frequently encountered.

Careful shaping of the external cowling lines to eliminate pressure peaks or local high velocities insures a smooth external air flow, which in a body of adequate fineness ratio gives rise to little drag. Avoidance of critical-flow regions in locating duct exits eliminates another common source of drag.

This particular installation is of added interest because of the ease with which other features fitted into a design primarily dictated by cooling considerations. Figure 3 shows this installation with the addition of the exhaust and intake system. The exhaust stacks which discharged below the cowling were shrouded as a protection against fire and to prevent heating of the cooling air by the hot stacks. Dual carburetors were fed from a manifold open at the front to the ram pressure entering the plenum chamber and connected in the rear to the exhaust shrouds which could be used as carburetor-air preheaters. Free swinging doors fitted with an override control permit connection of the carburetors either to the full ramming intake, or to the hot-air stoves. Backfire control was obtained through slamming of the free-swinging doors and relief through a backfire door at the bottom of the cowling.

Station-by-station analysis of the internal-flow system is a powerful aid in the preliminary design of power-plant installations, and is equally important in the evaluation of existing configurations. Such an analysis, based on flight tests of the installation under discussion, is illustrated in condensed form in figure 4. The method employed is described in detail in reference 8. In this type of analysis the system is divided into elements in each of which it is possible to state the significant changes of conditions of the air flow, and the condition of the air is traced step by step from the free stream to the point of discharge. The resulting information permits calculation of the drag force and power chargeable to the internal flow.

The first steps in a preliminary installation analysis are setting the physical conditions of pressure, temperature, and speed of the free stream, and obtaining the cooling air flow and pressure drop required by the engine. As the cooling air passes through the propeller a boost in total pressure is obtained under favorable conditions; lacking exact information it is conservative to neglect this boost.

The inlet area is calculated from the cooling air quantity and the desired inlet-velocity ratio. Ducting losses can be calculated from data in reference 9. Cooling-passage pressure losses are applied to calculate the total pressure at the duct exit, correcting, if necessary, for exit ducting losses, which however are usually small. From the dynamic pressure of discharge, which is the difference between total pressure within the exit and the external static pressure, the discharge velocity may be calculated. The necessary exit area is then obtained directly from the velocity and quantity of discharge.

Internal drag is calculated as the product of mass flow and the change in velocity from that of flight to that remaining when the pressure after discharge returns to that of the free stream. The calculations of figure 4 were made from correlated flight test data of the installation described and have an accuracy much better than could be obtained by estimate alone. The internal drag in the case illustrated is roughly 1 pound, corresponding to 1/3 horsepower at the 110-mile-per-hour flight speed. This low value for internal drag shows that the drag characteristic is well within tolerable limits.

Pusher aircraft, particularly of the seaplane type which may be required to taxi for extended periods at fairly high power need some means for augmenting the cooling air flow at low forward speeds. Where the geometry is favorable, cooling fans, discussed in references 10 and 11 offer the most obvious solution. In unsymmetrical installations such as that described herein, exhaust ejectors may be employed. A theory of exhaust ejectors is given in reference 12.

Other papers of value or interest to the installation engineer include reference 13 on cooling fin design; references 14 and 15 on cooling-duct exit flaps; reference 16, which gives the NACA cooling-correlation theory, by means of which engine-cooling air requirements are related to operating conditions; and, references 17 and 18 which present practical application of the theory to multi-cylinder engine installations.

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Rubert

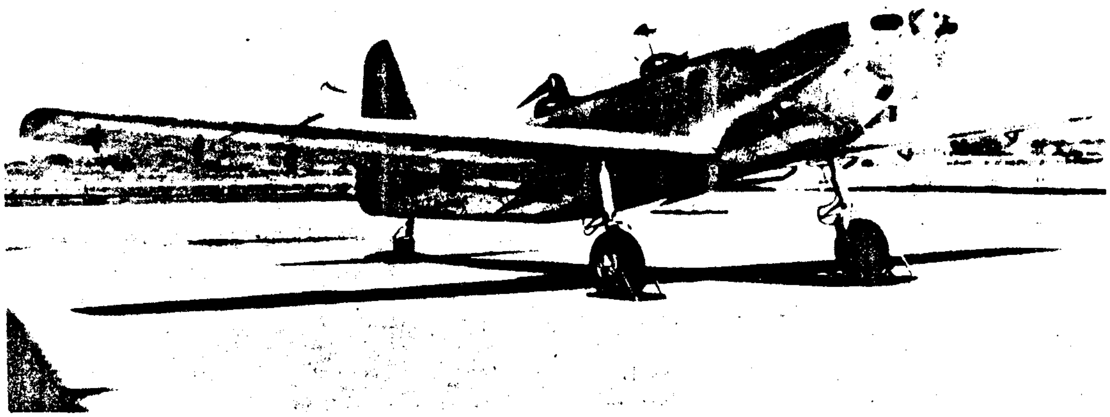


Figure 1.- Fleetwings model 33 trainer before installing target-airplane power-plant arrangement.

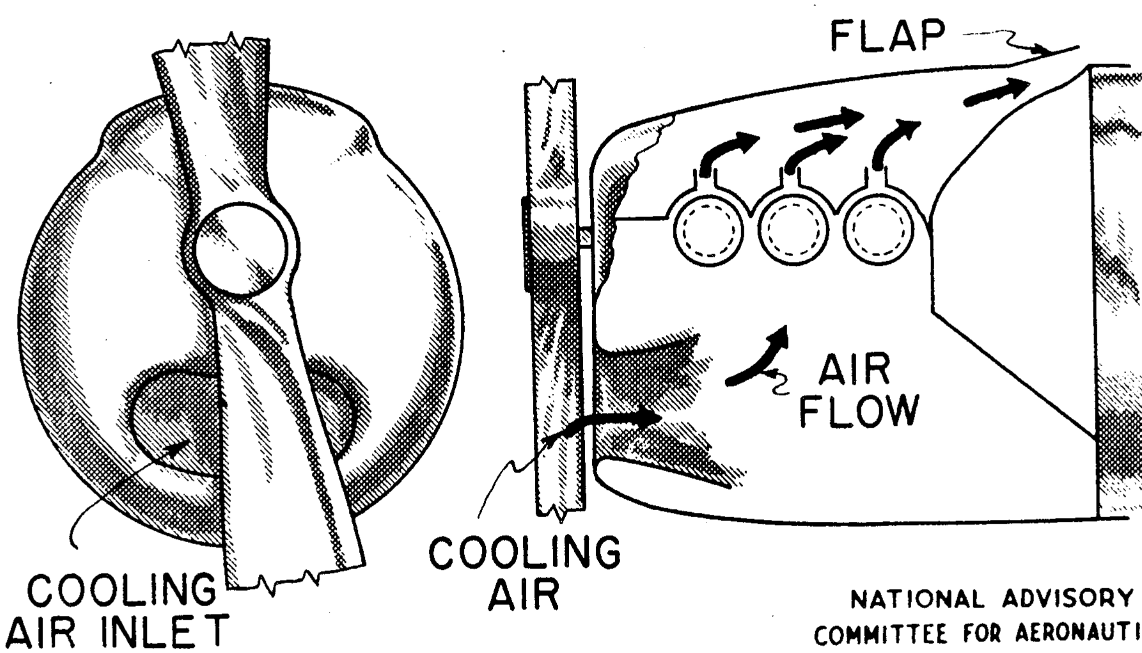


Figure 2.- Cooling arrangement for cowling as developed for use on target airplane.

Rubert

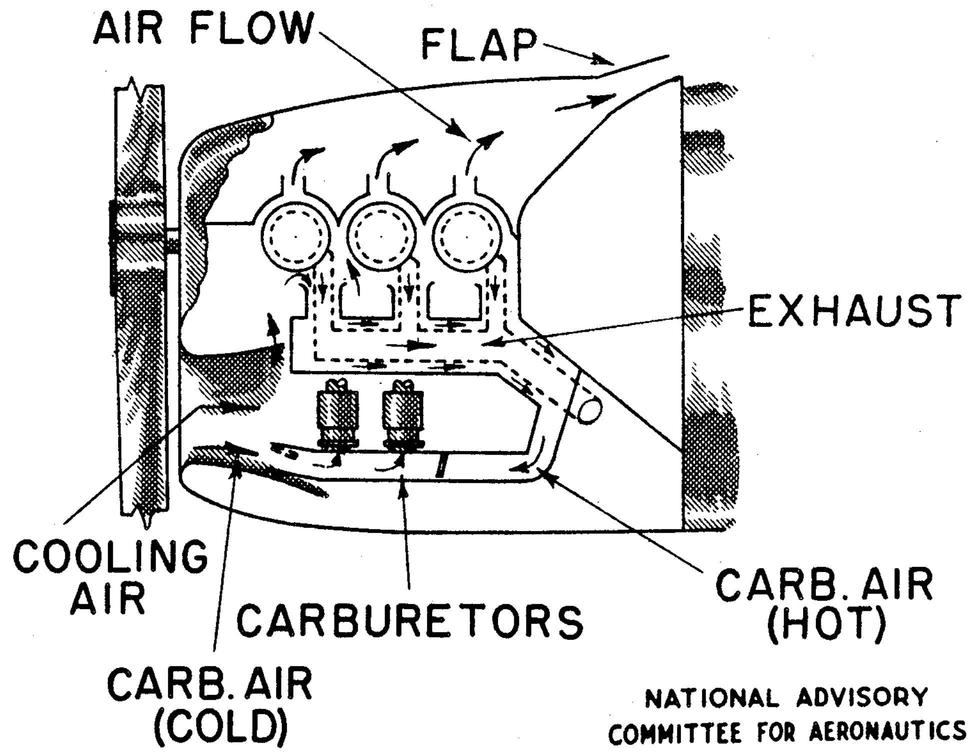
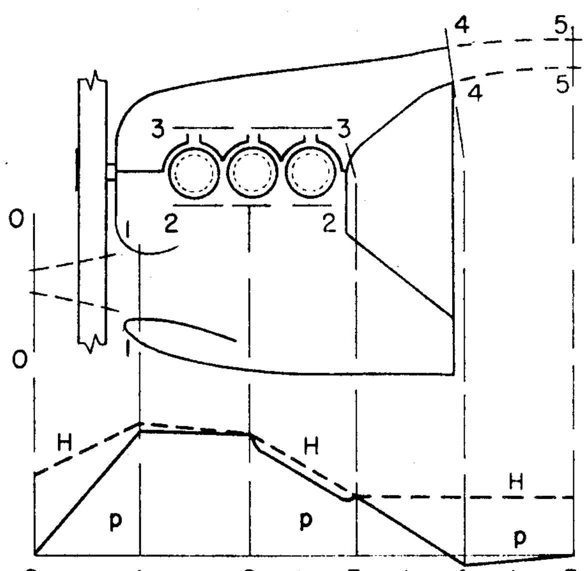


Figure 3.- Induction and exhaust system arrangement in target-airplane cowling.



	0	1	2	3	4	5
H, TOTAL PRESSURE, lb/sq ft	2070	2089	2084	2062	2062	2062
p, STATIC PRESSURE, lb/sq ft	2041	2086	2084	2062	2038	2041
V, VELOCITY, fps	165	41	–	–	156	149

	0–1	1–2	2–3	3–4	4–5
ΔH	+19	-4	-22	0	0
Δp	+45	-2	-22	-24	+3

Figure 4.- Internal-flow analysis.

ICING OF ENGINE INDUCTION SYSTEMS

By Willson H. Hunter

The need for better ice protection for light airplane induction systems was clearly reemphasized by reports that in 1945 there were over 195 forced landings of personal aircraft due to this form of icing, as compared with no forced landings of transport aircraft for the same cause despite their operation through more inclement weather. Analysis indicates that this striking difference in performance is attributable partly to inherent differences in icing characteristics of the two induction systems and partly to differences in skill and training between private and transport pilots. First, let us examine the differences in icing characteristics.

Most conventional induction systems contain the same basic elements but differ significantly in the manner of admitting fuel to the system. We will first consider a typical light airplane induction system in which the fuel feed is caused by venturi suction. The basic elements of this system comprise: a cold air intake; a carburetor which meters the air, proportions the fuel spray and throttles the air flow; and a manifold for conveying the fuel-air mixture to the engine cylinders.

Three distinct types of ice can form in this system: impact ice, throttling ice, and fuel-evaporation ice. Impact ice will form whenever visible water droplets in air below 32° F impinge and freeze on cold surfaces at the air intake, inside the duct, and on the carburetor parts. The rate of impact ice formation will increase with airspeed and water content, will be most prevalent above 15° F, but can be expected at temperatures well below 0° F. Any air filter or screen in the cold air inlet will quickly become blocked under these conditions, but the pilot is immediately warned by the presence of ice elsewhere on the aircraft. The private pilot will usually take evasive action under these circumstances. Dry snow and sleet present a similar problem in the case of the filter but are not generally serious otherwise.

Throttling ice forms whenever the effective temperature drop due to the expansion of moist air through the carburetor venturi and through a partly closed throttle is enough to condense water and freeze it at or beyond the throttles. Analysis and experiments show that this phenomenon can occur in clear air but is limited to inlet air temperatures between 32° F and 40° F. Throttling ice can form in any system in which metal surfaces downstream of the point of minimum pressure are cooled below freezing. The throttling process is most important, however, because it contributes at all temperatures to the more prevalent and serious fuel-evaporation icing process.

Fuel-evaporation ice forms whenever the air and adjacent metal surfaces are cooled below freezing due to the evaporation of fuel. When this occurs additional water condenses from the air and freezes, together with any ingested rain. Since a finite time is required to evaporate fuel, it is to be expected that in a rapidly moving air stream serious cooling would develop only well beyond the fuel spray. In a well designed system operating at full throttle this is actually the case, but at part throttle, in the system we are considering, fuel impinges directly on the throttle plate and eddies in the turbulent region beyond the throttle in a manner that both increases and localizes the refrigeration effect. At very low air flow or high water concentrations, ice may form on the fuel nozzle itself sufficiently to obstruct fuel flow. It has been learned that fuel-evaporation icing is aggravated by more volatile fuels but is not significently affected by increase in fuel-air ratio or by fuel temperature. Our experiments indicate that in many induction systems fuel-evaporation ice occurs in clear air at inlet temperatures as high as 100° F. Thus it can be seen that induction system icing is a year-around problem and is not necessarily associated with bad weather, or clouds, or any particular geographic area. This explains why inexperienced pilots are so easily fooled by induction-system icing.

The three forms of icing will all produce a loss in engine air flow at a given throttle setting, but they may also seriously affect fuel metering, freeze the throttle, or upset the even distribution of mixture to the cylinders. Large amounts of impact and fuel-evaporation ice before or after the carburetor will cause little effect on air flow at part throttle because the system is then handling only a fraction of the design air flow. However, small amounts of ice on the throttle edges will quickly obstruct air flow and may freeze the throttles.

During cruising flight at constant altitude, a loss of air flow will be detected as a loss of engine rpm and airplane speed. An alert pilot who does not suspect what is happening will have a tendency to advance the throttle to regain power. This process can continue intermittently until the throttles stick or until full throttle is attained, but cruising power may continue to be lost until forced landing is necessary.

With the natural aspirating carburetor we have been discussing, ice may have interfered with the fuel feed, during the previous example, until rough or erratic engine operation resulted in backfiring or complete engine stoppage. In some cases pilots have recommended leaning out the mixture excessively to force backfiring to blast ice formations loose. This practice is too haphazardous and dangerous from other standpoints to merit any consideration as a means of ice protection.

When an engine is idling or being operated on the ground for long periods at low power, the induction system can become loaded with ice and later fail to develop full throttle power during take-off because ice is throttling the air flow. During a long glide, if the engine is throttled back and held at or near idling, it can become iced and cooled until complete engine stoppage occurs and power can no longer be recovered. These are conditions that confront the inexperienced pilot.

The common remedy for induction-system icing, available in all aircraft, is for the pilot to exercise judgment at the proper time to de-ice the system and prevent further icing by manually valving to an alternate source of air which is heated by an exhaust shroud. If the valve is not frozen with impact ice and can successfully shut off the cold ram air, and if the engine is still developing enough power to provide sufficient exhaust heat, the system will be quite effectively protected after about 30 seconds. One defect of this method of protection , aside from the fact that the pilot must remember to turn it on and off, is that full hot air in sea-level engines may lengthen take-off distance and reduce climb dangerously for short runway operation. Coupling the hot-air control to the throttle so heat would be applied automatically at low power, yet could be manually selected when needed would help a great deal. But this is not enough; better means of protection are needed for personal aircraft.

If fuel is injected under pressure at a point downstream from the throttle, the refrigeration effect of fuel-evaporation in the carburetor body and on the throttle plate will be considerably reduced and the greater injection pressure and shielding of the fuel nozzle will help prevent stoppage of fuel flow. It will be apparent, however, that impact ice and throttling ice are not at all prevented by this change in fuel injection, although the seriousness of their effects on air flow and fuel metering is greatly reduced. In essence, this represents the pressure carburetor which has been standard equipment on transport aircraft for many years and was recently made available in smaller sizes for light aircraft. Whereas 150° F air is specified for the protection of natural aspirating carburetors, only 130° F air is required for the pressure carburetor in this form.

Some transport engines now embody a further improvement in pressure-carburetor fuel injection. The fuel nozzle is removed entirely from the vicinity of the carburetor and fuel is admitted through a spinner or through the impeller directly into the supercharger. Since the supercharger is heated by the compression of air, fuel evaporation is entirely eliminated. A radical reduction in heat requirements is thus made possible and the engine is free from icing above 40° F. Unfortunately, sea-level engines cannot take advantage of this improvement until means are provided for properly proportioning the fuel among the several intake pipes.

A still further improvement in ice prevention can be accomplished by utilizing means other than the carburetor for metering and then injecting the fuel into or near the engine cylinders. The speed-density metering system developed by the Royal Aircraft Establishment, Farnborough, is such a system and is currently being fitted to some rather small engines. Eliminating the venturi air meter reduces the seriousness of impact ice and eliminates one cause of throttling ice. The heat requirements for this system are still lower but ice on the throttles will still require heating them, a job the British are doing with internal hot-oil circulation. It is gratifying to learn that a new simplified form of full fuel injection is now available in this country and has already been adopted as standard equipment on four makes of light aircraft. This is a long and commendable step towards making safe flying easier for the private pilot.

The complete simplification of the induction-system icing problem lies in excluding water from the engine air intake in combination with the last mentioned fuel-injection system. Water separation by means of judicious air scoop design has already been investigated by NACA as will be explained later and research continues along these lines.

In 1941 the NACA began a broad investigation of induction system icing at the Bureau of Standards and continued the research at the NACA Cleveland Laboratory in late 1943. Some of the results are now available as declassified Advance Restricted Reports while the majority of the results, obtained for the Army Air Forces, are expected to be issued soon as Wartime Reports and are also being summarized in a Technical Note. In addition a 16 millimeter sound motion picture for pilot information was prepared through the joint efforts of NACA and the Air Materiel Command showing induction system icing and de-icing as seen through special transparent carburetor and engine parts. It is hoped that this film may soon be declassified and made generally available for pilot information.

For the purposes of this paper several figures have been prepared to illustrate typical research results. Figure 1 shows the benefits of water separation in preventing impact ice within the induction system of a large twin-engine cargo aircraft. The top left illustration shows the relative paths of water and air into the standard scoop entrance; next the resulting impact ice obtained during tests in the Cleveland Icing Research Tunnel; in the bottom photograph ice almost completely blocks the carburetor screen about 5 feet downstream of the intake. On the top right, water and air streamlines are shown for an experimental form of undercowling scoop; next, typical impact icing on the exterior cowling surface; and, bottom, a complete absence of screen ice under identical test conditions. Ninety-five percent of the freezing water was excluded by this design with no loss of ram in cruising flight conditions for critical altitude. This is the first step toward a truly sheltered, water-excluding air intake.

Figure 2 illustrates typical serious impact icing, with and without the carburetor screen, of a three-barrel pressure carburetor with spinner injection.

Figure 3 shows serious throttling and fuel-evaporation ice in the supercharger inlet elbow (top photos) and underside of a two-barrel pressure carburetor (bottom photos) for three test conditions. The two pictures at the right illustrate the result of continually opening the throttles to maintain power during icing conditions; the throttles finally froze in the position shown and could not be returned to normal when a large plug of ice broke loose and permitted air flow in excess of cruise power.

The limiting icing conditions for this engine and pressure carburetor are shown in figure 4, for three power settings. Carburetor air temperature is plotted vertically and water content of the air, horizontally. To the left of the central vertical line are clear-air conditions; to the right are cloud and rain conditions (simulated rain injections in carburetor air inlet in grams per minute). It is important to note that ice was observed for clear air conditions at inlet temperatures of 100° F. Increasing power by opening the throttles generally reduces the temperature at which ice will form in the system. Throttle opening has little effect on the icing limits with large amounts of water present.

If you will disregard the complexity of figure 5 you will note that carburetor air temperature (C.A.T.) is again plotted vertically and water-air ratio horizontally. The curved lines sloping upward from left to right represent constant percentages of relative humidity (R.H.) from 10 to 100; clear air conditions are to the left of the 100-percent line, cloud and rain to the right. The lines sloping downward from left to right are lines of constant enthalpy H. The six heavy curves represent the upper temperature limits for the occurrence of serious induction system icing at cruise power for six different engine carburetor combinations. It is important to note that four of the engines exhibit serious icing tendencies in clear-air conditions. The impressive improvement from the topmost curve to the lowest curve is accounted for by the use of spinner fuel injection in the latter case.

Despite the general seriousness of the icing characteristics of these military and transport induction systems it must be realized that highly trained pilots, adequate carburetor heat, alcohol sprays for emergency de-icing, and good instrumentation insure safe operation. It would not be reasonable to expect personal aircraft to be operated as safely with this same equipment because of the relative inexperience of the average private pilot.

If you will permit, I should like to review a precedent established by another personal vehicle which has somewhat similar problems. The dictates of freedom from abrasive wear, operating quietness, and freedom from backfire dangers, long ago caused the automobile manufacturers to adopt the now familiar air-cleaner, silence and backfire screen combination. All-weather usage of automobiles made it good practice to locate the air intake in the warm sheltered area close to the top of the hood. Ease of starting with low volatility fuels and freedom from the abuse or forgetfulness of inexperienced drivers led to the general adoption of the automatic choke and thermostatically controlled hotspot. Even though first cost still dictates the use of natural aspirating carburetors the bad icing qualities have been adequately offset by the automatic choke and hotspot which prevent serious icing during cold starting and at low engine power. We all realize that the present day automobile induction system does permit trouble-free operation of engines, comparable in size to those in most light airplanes, by personnel possessing far less skill and judgment than commonly expected of private pilots. This example, demonstrated by millions of automobiles, is worthy of our careful consideration.

I do not hold up the automobile induction system necessarily as a model to copy. The elements of such a system are all at hand and the principles are understood. It is possible that a very satisfactory induction system for personal aircraft will contain the following features:

(1) A single, warm, sheltered air intake in a rammed plenum chamber.

(2) An air-cleaner unit which also serves to silence the air intake and to extinguish backfires.

(3) Some type of speed density metering device replacing the carburetor.

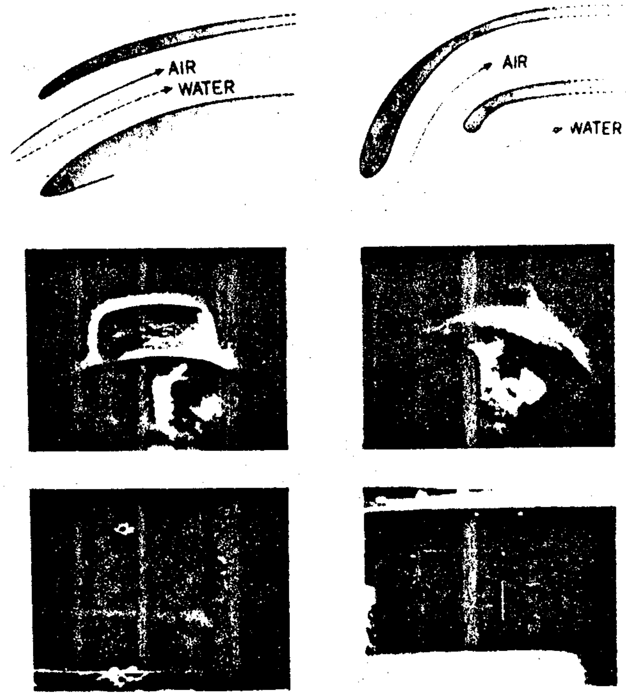
(4) Fuel injection at each inlet valve.

(5) A simple throttle body integral with the engine and warmed by it.

(6) A thermostatically controlled hotspot to insure good fuel vaporization at low air temperatures and to eliminate need for pilot attention to induction-system icing.

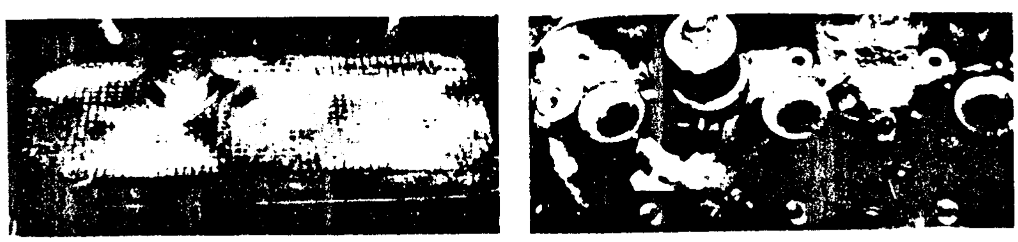
The reward for such a design will be safer flying and reduced training time for many future pilots and an important step will have been taken toward the goal of the all-weather personal aircraft.

Hunter', Willson



STANDARD UNDERCOWL

Figure 1.- Effect of scoop design on carburetor screen icing.



C.A.T., 29°F; R.H., 100%; simulated rain injection
500 G/M; air-flow reduction, 50% in 6 minutes

C.A.T., 31°F; R.H., 100%; no simulated rain injection,
air-flow reduction, 26% in 15 minutes

Figure 2.- Impact icing in P. & W. R-2800-51 induction system at low cruise power.

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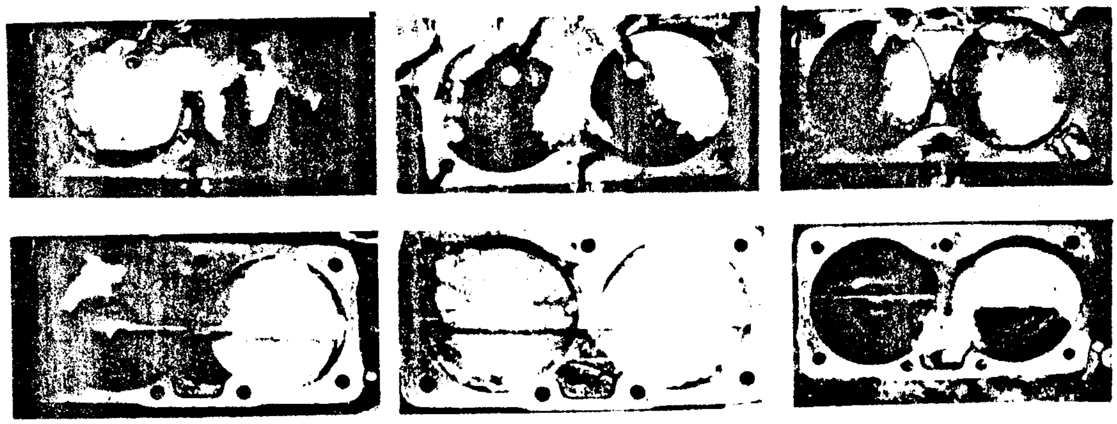

C.A.T., 40°F
R.H., 100%
Water, 350 G/M

C.A.T., 37°F
R.H., 100%
Water, 650 G/M

C.A.T., 40°F
R.H., 100%
Water, 0 G/M

60 % cruise power

Figure 3.- Typical icing in V-1710-89 induction system.

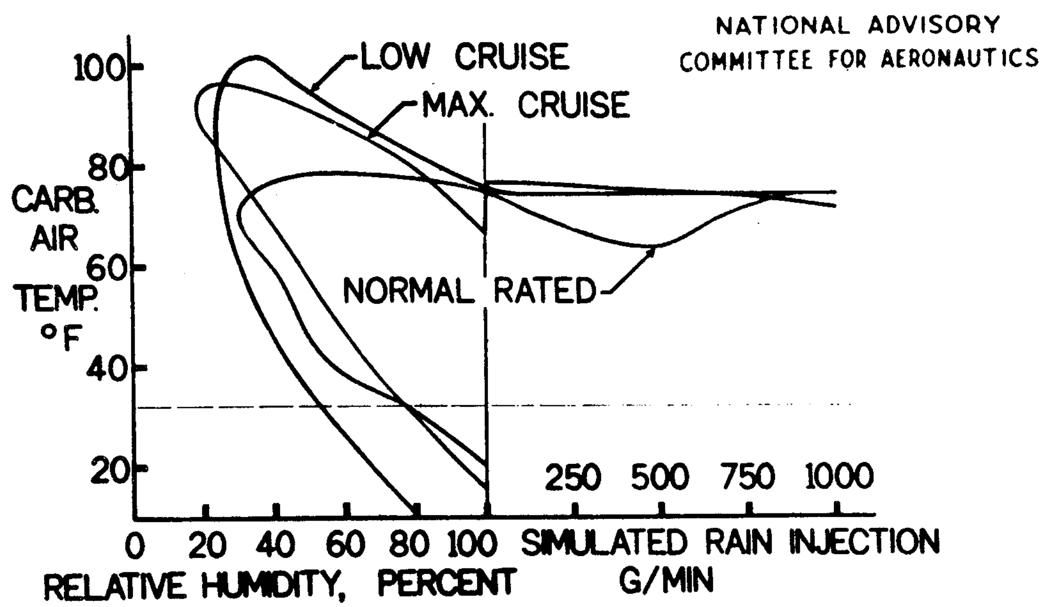

Figure 4.- Limits of visible icing in Allison V-1710-89 inlet elbow and carburetor with AN-F-22 fuel.

Hunter, Willson

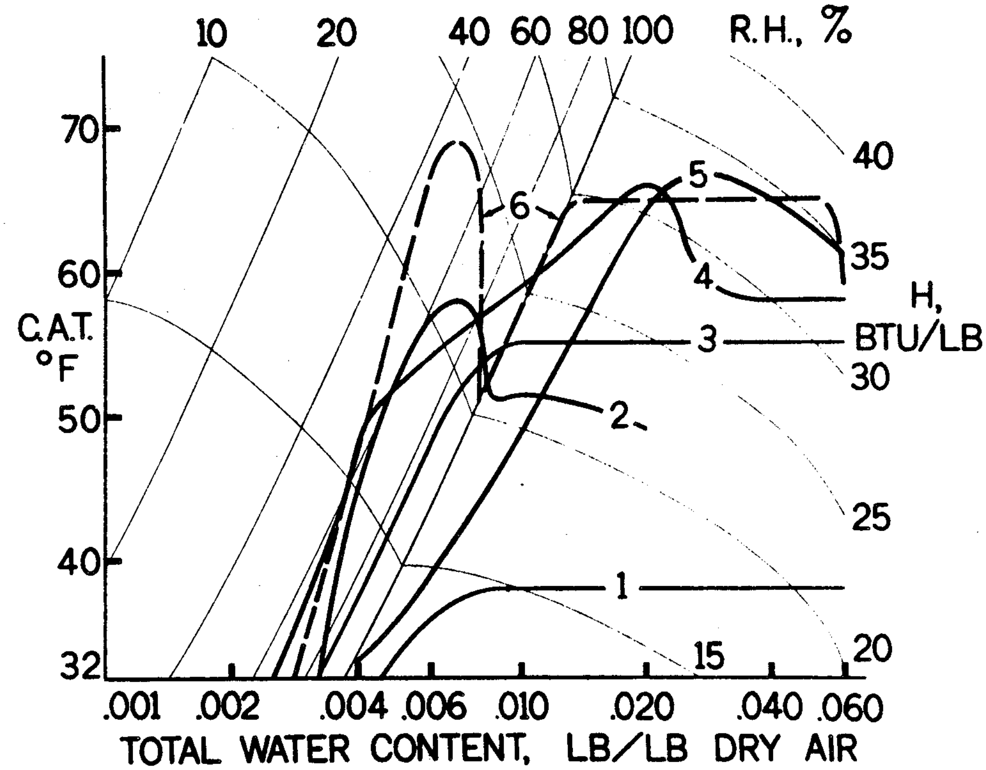

Figure 5.

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PROSPECTS FOR NEW TYPES OF PROPULSIVE SYSTEMS FOR PERSONAL AIRCRAFT

By John C. Sanders

The dominance of the reciprocating engine as a power plant for aircraft has been threatened in the past few years by newer types of propulsive systems. The new turbojet engine is less complicated than the large reciprocating engines, and under high speed flight conditions this new engine has a lower ratio of weight to thrust. Since the qualities of light weight and simplicity are sought for personal airplanes, a preliminary review of the characteristics of some of these new engines has been made to provide a basis for a scientific guess as to whether research on the application of them to personal aircraft should be conducted. Consequently, the conclusions of this study are not directed toward recommendations for the adoption of the new engines.

The list, which is far from exhaustive, of the engines considered in this preliminary study, includes the turbojet engine, the buzz-bomb engine, an application of the ram jet, and a gas turbine driving a propeller. No consideration is given to the possibility of improving the reciprocating engine.

The study gives an analysis of engine performance but neglects other important considerations such as cost, safety, and noise. The performance characteristics considered are fuel consumption, thrust-weight ratio, aircraft range, and maximum speed. A basis for comparison is achieved by specifying that the several engines must have the same static thrust equal to that of a selected reciprocating engine and propeller. These engines are considered applied to an airplane weighing 1200 pounds, and having cruising and maximum speeds of 95 and 105 miles per hour with a reciprocating engine.

Figure 1 shows a comparison of the specific fuel consumptions. The reciprocating engine is shown to have a lower fuel consumption than any of the other engines. This virtue is likely to hold this engine in a position of great importance for a long time. The explosion ram jet has a ridiculous fuel consumption of over 12 pounds per horsepower hour as compared with 0.7 pounds per horsepower hour for the reciprocating engine. The turbojet engine also has a very high fuel consumption. The gas turbine driving a propeller has a fuel consumption of 1.3 pounds per horsepower hour. This figure was estimated from component efficiencies observed in tests of the small turbojet engine considered in this study. Improvement of the efficiencies of these components may eventually reduce the fuel consumption to 0.8 pounds per horsepower hour.

Artists' impressions of how the turbojet engine and gas turbine driving a propeller would be installed in a personal airplane are shown in figures 2 and 3.

Discussion of the ram-jet engine has been postponed until now because the characteristics of this engine must be considered in some detail. Figure 4 shows the effect of airspeed, expressed as Mach number, on the fuel consumption. At a Mach number of 0.13, which corresponds to a flight speed of 100 miles per hour the theoretical fuel consumption is shown to be at the very high figure of 80 pounds per horsepower hour. Reasonable fuel consumptions will be achieved only at speeds above a Mach number of 0.7.

A plan for utilizing the ram jet on slow-speed aircraft which has been suggested frequently is to attach ram jets to the tips of the propeller blades. Then Mach numbers above 0.7 could be realized at the ram jet with the airplane traveling at only 100 miles per hour. An engine embodying a similar idea is shown in figure 5. In this engine air is induced into the hub and pumped by centrifugal action through hollow blades where heat is applied by combustion, and the hot air is ejected from the blade tips. Estimates of the performance of such an engine, accounting for burner and windage losses, indicate that a fuel consumption of 3 pounds per horsepower hour might be expected. This value is compared with the fuel consumptions of the other engines in figure 1 and is shown to be much higher than for the reciprocating engine.

A comparison of the ratio of weight to static thrust shows a different order of merit than found in the study of fuel consumptions. In this comparison shown in figure 6 the static thrusts of the several engines are equal. The reciprocating engine with its propeller is shown to be equally as heavy as the explosion ram jet (or buzz-bomb engine), and that these are the two heaviest engines considered. The turbojet engine is somewhat lighter and the jet-operated propeller is much lighter than the reciprocating engine The weight of the turbojet engine was obtained from the actual weighing of a turbojet engine of the size used in this analysis. The buzz-bomb weight was obtained from the thrust-weight ratio of a buzz-bomb motor. The weight of the gas turbine and propeller was estimated from the weight of the turbojet engine, allowing for the weight of gearing and propeller.

The two characteristics just considered of fuel consumption and engine weight may be reduced to a common term by computing the weight of the engine plus the fuel required for specified flight distances. Figure 7 shows the results of such computations. In this figure the abscissa is the range for which the airplane is designed to fly and the ordinate is the weight at take-off of the entire power system, including engine, propeller, accessories, and fuel tanks with fuel. In the case of the reciprocating engine it may be seen that if the airplane were designed to fly 300 miles and had a full load of fuel on board, the weight of the entire power system would be 240 pounds. The weight-range characteristics of the explosion ram-jet engine are shown in the top curve. Little can be said to recommend this type of engine on a weight-range basis because the initial weight is approximately equivalent to that of the other engines and the fuel load required is extremely high. In fact, for ranges of 300 miles or greater, the weight of the power plant would considerably exceed the weight of the remainder of the airplane. The turbojet engine is shown to be better than the explosion ram-jet engine but is heavier than the reciprocating engine.

The low weight of the jet-operated propeller without fuel opens a possibility for its use in spite of its high fuel consumption. It may be seen that at a range of 150 miles the weight of the jet-operated propeller and its fuel would be no greater than that of the reciprocating engine and for shorter flights the jet propeller would be lighter. The gas-turbine driven propeller is here again shown to be a close competitor to the reciprocating engine. The same qualifications as to the accuracy of these calculations must be made as before. The performance of these two power plants is estimated to be sufficiently near equality that no significant differences should be pointed out from this figure.

The comparisons so far made have been quite unfavorable to jet propulsion and the source of the difficulties have arisen from the very low static thrust and high fuel consumptions at flight speeds below 100 miles per hour. When, however, jet-propulsion engines are used which produce the same static thrust as the propeller engines, the power available at cruising speed with the jet engines is considerably greater. Thus the maximum flying speeds of the airplanes equipped with the jet engines herein described would be greater than for the propeller-driven aircraft. Figure 8 shows the estimated maximum flight speeds using these engines. It may be seen that the explosion-jet propulsion engine would achieve a flight speed of 160 miles per hour and the turbojet engine would achieve 142 miles per hour as compared to 105 miles per hour for the three propeller engines.

In casting up the accounts for the several engines discussed, it appears that where range is a factor the reciprocating engine will be as good as any of the others. Its chief competitor appears to be the gas-turbine driven propeller and the jet-operated propeller. The jet-operated propeller will not be useful for long-range flights but its simplicity will tend to make it a cheaper engine, thus meeting one of the requirements of the future engines. The gas-turbine engine appears to be a close competitor in weight and fuel-consumption characteristics but I have no idea what the comparative cost would be. Jet propulsion might be useful for high-speed airplanes with short ranges.

In closing, I would like to caution you that these figures are estimates made for the purpose of determining whether research on any of these engines should be considered for the purpose of encouraging use of these engines in light aircraft. The study indicates that the gas-turbine driven propeller and the jet propeller warrant further study and research. Such a recommendation, however, is made only to research organizations, since the undertaking of the development of these engines at present would be extremely costly and further research may reveal the impracticality of applying them to the personal airplane.

SYMBOLS

C_D	drag coefficient D/qS
D	drag, pound
q	dynamic pressure, pounds per square foot
S	wing area, square feet

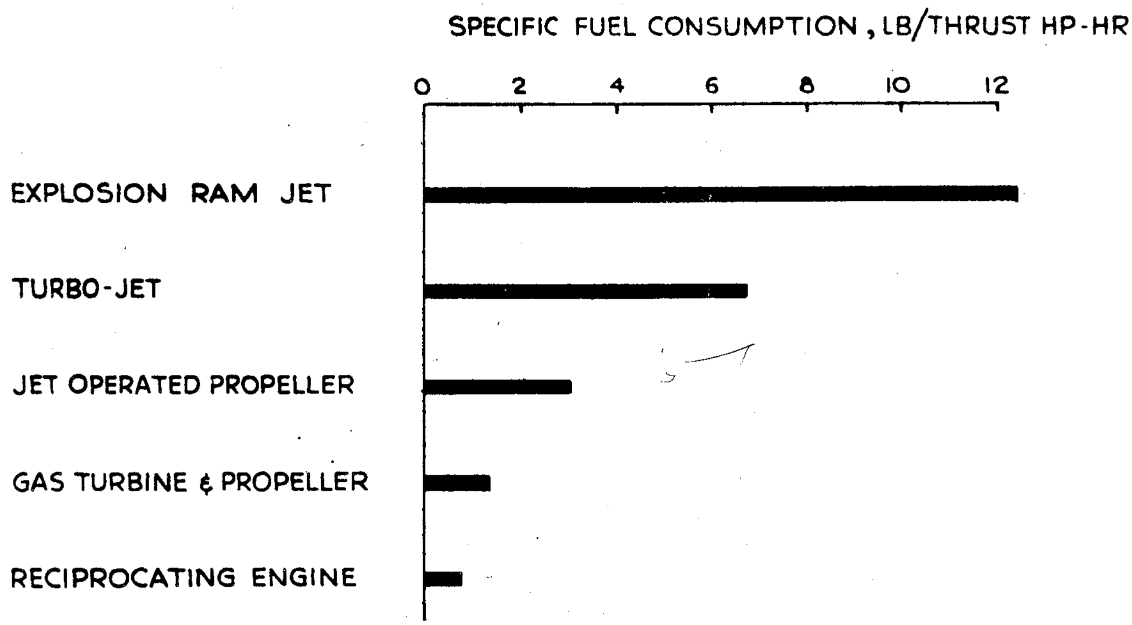

Figure 1.- Representative fuel consumptions of several power systems. Airspeed, 100 miles per hour.

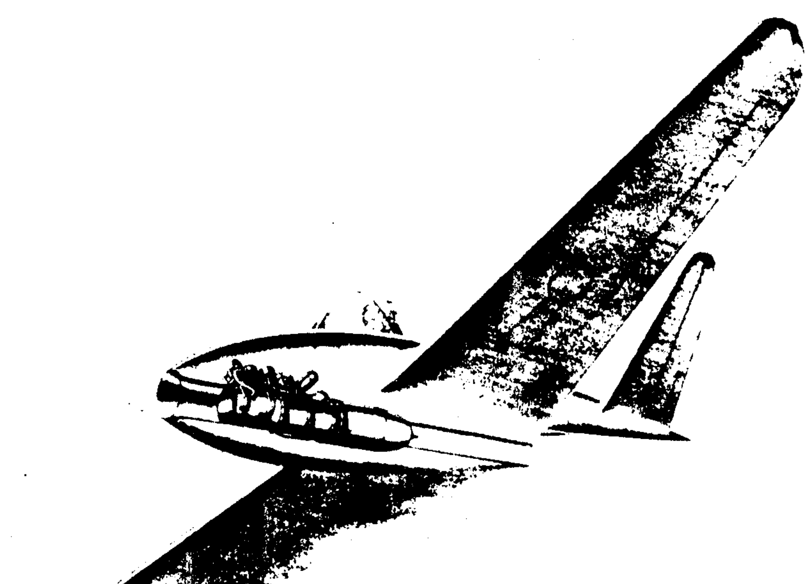

Figure 2.- Turbojet-propelled airplane.

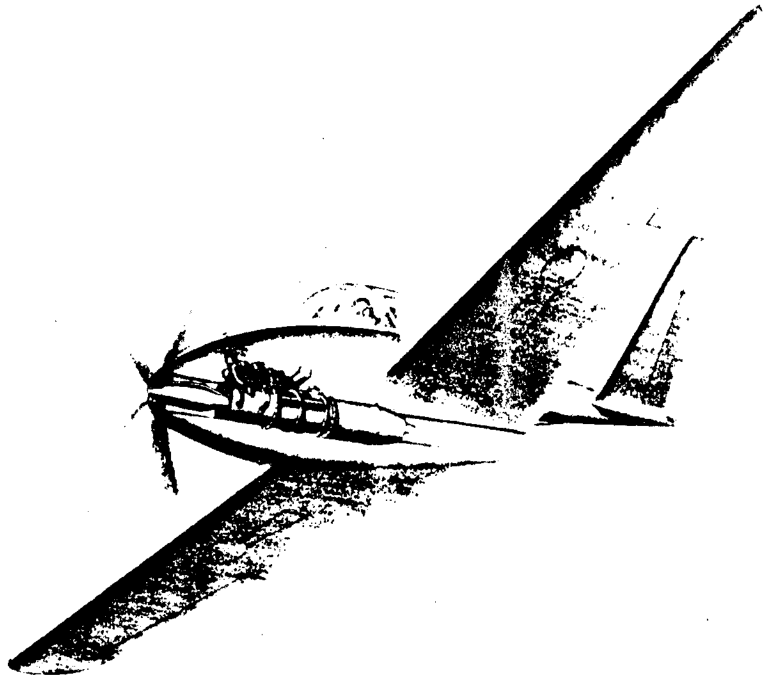

Figure 3.- Gas turbine and propeller arrangement.

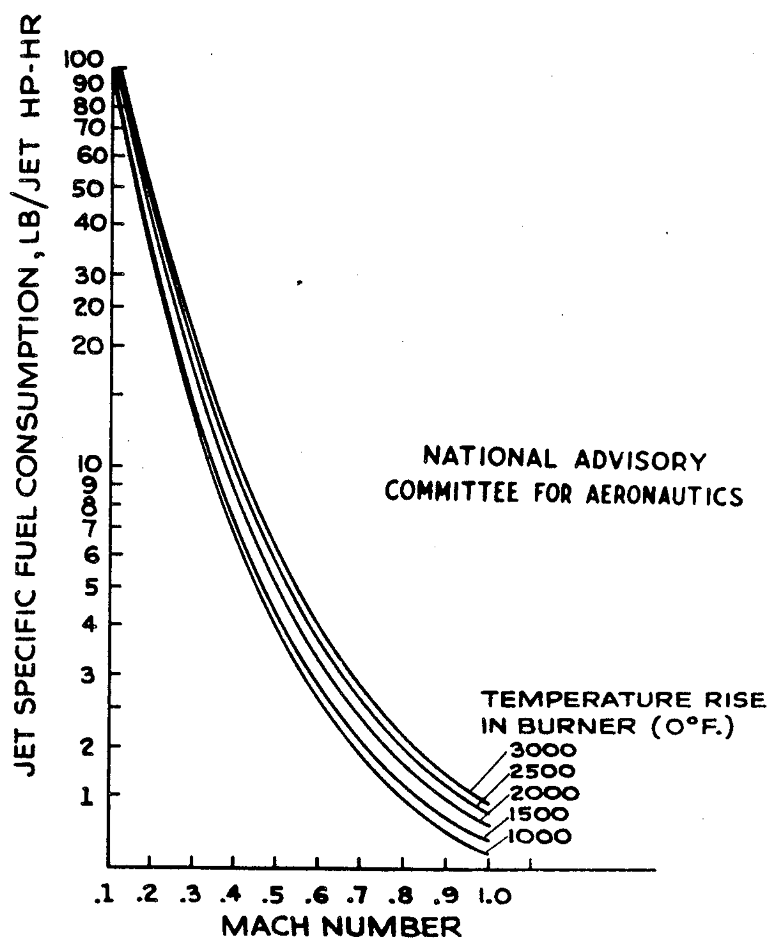

Figure 4.- Steady flow ram-jet fuel consumption.

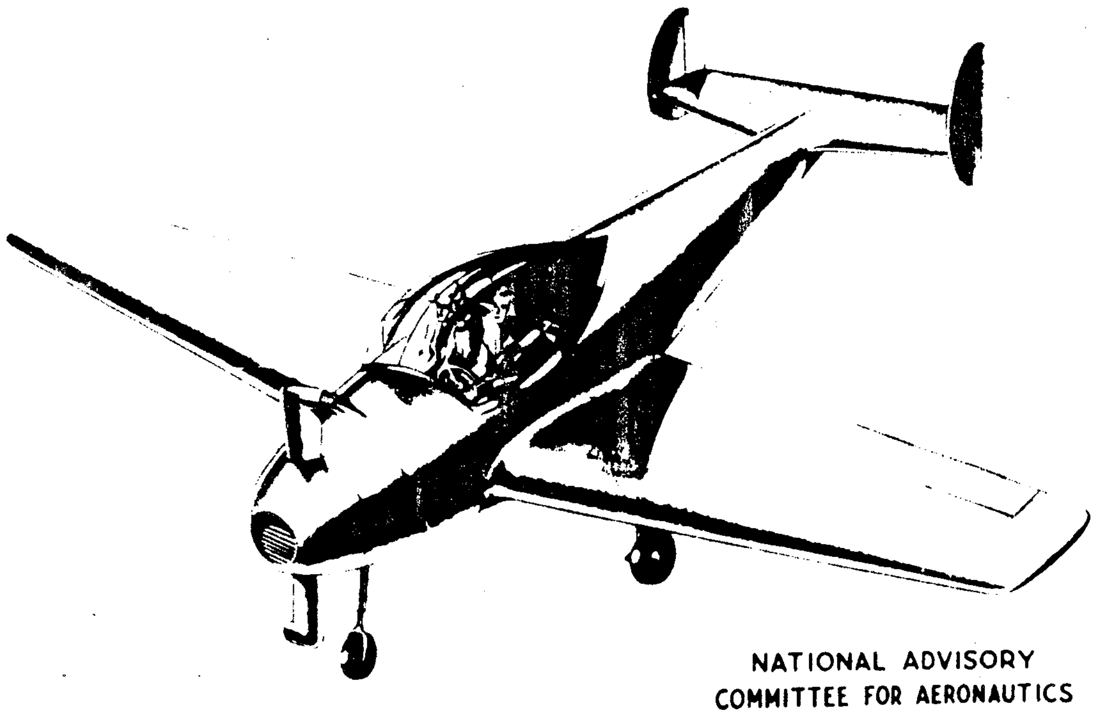

Figure 5.- Jet-operated propeller installation.

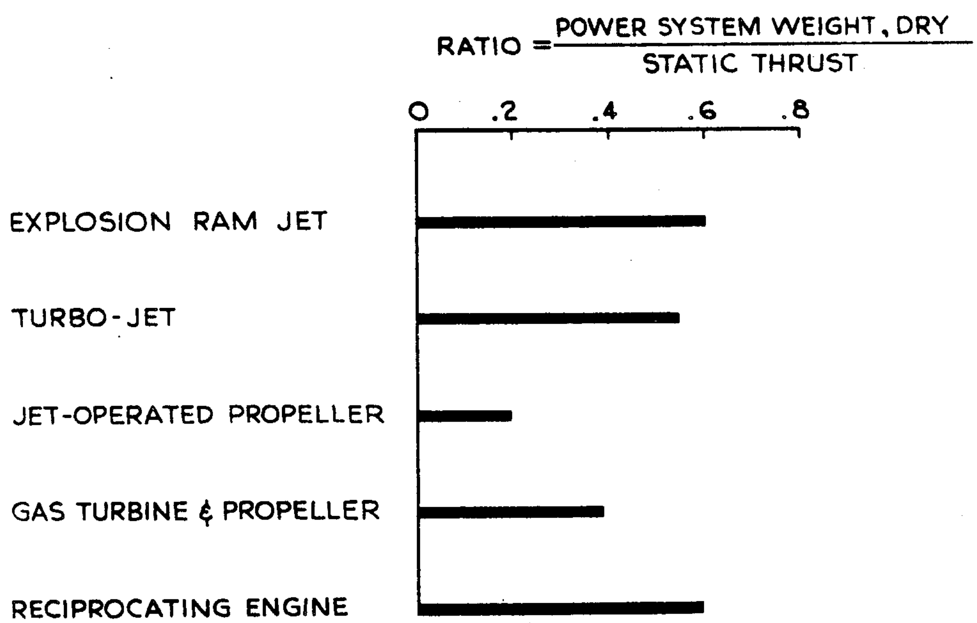

Figure 6.- Ratio of power system weight to static thrust.

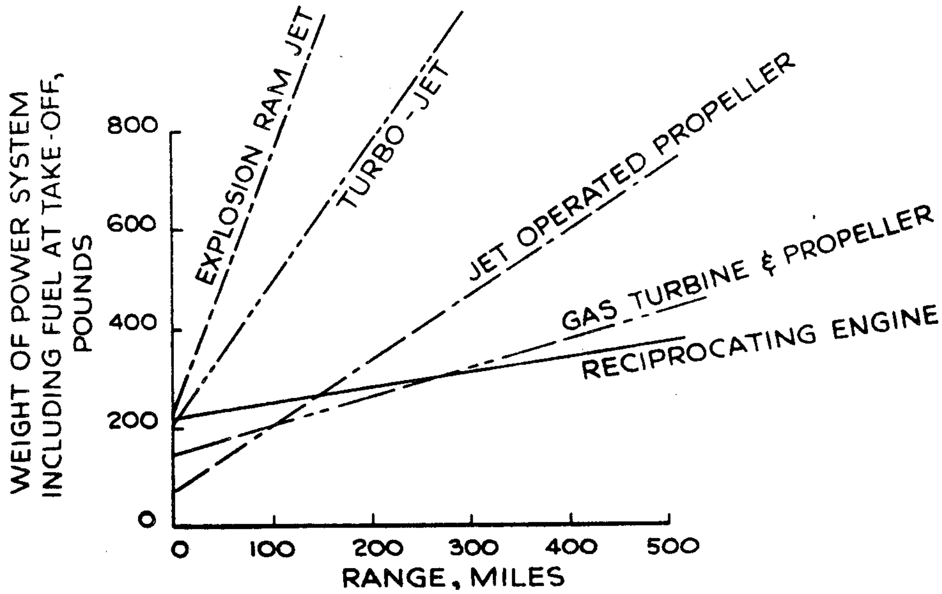

Figure 7.- Range-weight comparison of several power systems. Static thrust, 372 pounds; cruising drag, 153 pounds; cruising speed, 95 miles per hour.

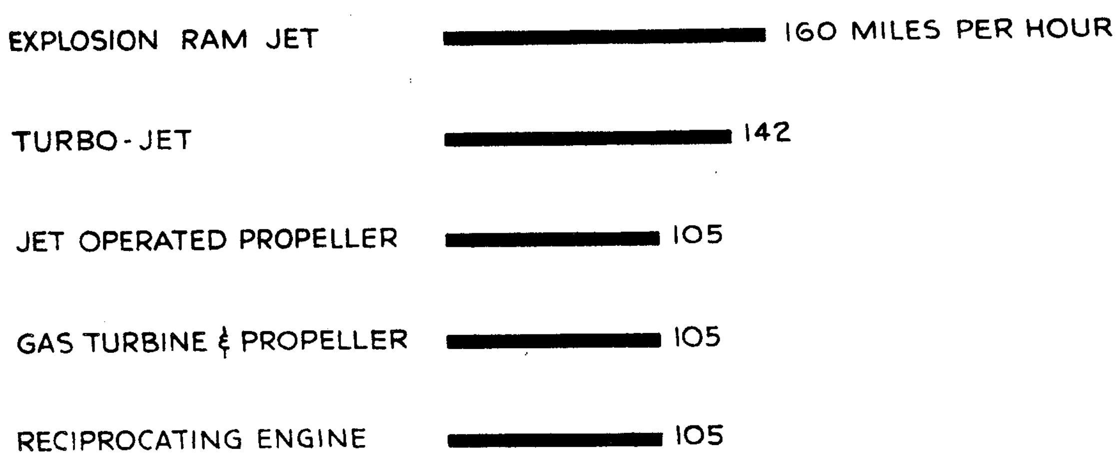

Figure 8.- Maximum speeds of an airplane with several types of power systems when all power systems produce the same static thrust. Equivalent value of C_DS, 6.59 square feet; static thrust, 372 pounds.

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NEW RESEARCH

CURRENT AND PROPOSED NACA RESEARCH INVESTIGATIONS OF INTEREST IN PERSONAL-AIRCRAFT DESIGN

By Harrison C. Chandler, Jr.

INTRODUCTION

The NACA has received from several industry and Government organizations a large number of suggestions for research on a wide variety of personal-aircraft problems. In our study of these research suggestions we have been interested, first of all, in determining the extent to which NACA wartime research, results of which are now becoming generally available, will aid in providing solutions to the problems submitted. During this conference it has been shown that a considerable amount of the Committee's research on military-aircraft problems is of direct value in the design of personal aircraft.

Next, we have examined our current and contemplated research in various fields to determine which of these projects are applicable to personal-aircraft problems, and in some cases the scope of the projects has been extended to make the results of the research more directly applicable. In addition, several investigations have been initiated by the Committee's laboratories specifically to study problems which are peculiar to the light airplane field. I will discuss very briefly, first, the current research in various fields which is of interest in this connection, and, later, will turn to the projects aimed directly at the personal airplane.

RESEARCH HAVING APPLICATION TO PERSONAL-AIRCRAFT DESIGN

Stability

In view of the large number of variables that can be changed in designing the airplane for good stability characteristics, both power off and power on, it is extremely difficult to arrive, by analytical means, at the more favorable configurations for stability. The Langley Laboratory is, therefore, making a collection of the data of the many powered wind-tunnel models tested during the war. These data will be analyzed to determine the better configurations from the standpoint of longitudinal stability and also to determine the significant factors that contribute to good stability characteristics. A similar study of lateral stability is also being made, with a large part of the work, in this case, being done by the Ames Laboratory. One of the portions of the Ames Laboratory's lateral stability research is directed toward the problem of rolling moment due to sideslip, taking into account the effects of flaps, wing dihedral, engine power, and other contributing factors.

At the Ames Laboratory a project allied to stability has been initiated to provide a design basis for glide-path-control devices and flaps having minimum effect on airplane trim. This problem is regarded as particularly important for personal aircraft from the standpoints of both safety and utility.

As a part of its research program on stability problems the Langley Laboratory has made some specific tests dealing with (1) the effects of tail length, (2) the effects of engine tilt and skew, (3) the effects of wing position, and (4) the effects of high-lift flaps on longitudinal and lateral stability, with particular attention to the changes in stability that occur when power is applied. These projects are being reported individually.

In order to assist in reaching an understanding of power effects on dynamic longitudinal stability the Langley free-flight tunnel is conducting a flight study with a representative airplane model.

Wing Design

The Langley low-turbulence tunnel is extending its investigation of the characteristics of airfoils to a low range of Reynolds numbers in order to provide design data which will be directly applicable to personal aircraft over their entire speed range. An investigation also is under way in this tunnel to provide research data for airfoils fitted with a wide variety of types of high-lift devices. In addition to this research, work is being carried out in the low-turbulence tunnel on boundary-layer control as a means for extending the maximum lift of airfoils. As you know, high-lift is an important problem in the design of light airplanes.

Controls

Allied with the problem of obtaining high lift is the problem of providing suitable lateral-control devices to be used in conjunction with the high-lift devices. Spoilers, as Mr. Toll mentioned, either alone or in combination with the so-called "feeler" ailerons, appear to be the best solution for this problem. One current study deals with the effect of length and spanwise locations of spoilers.

Propellers

Mr. Crigler has outlined in his paper the results of recent NACA research which are regarded as most suitable for the selection of propellers of high efficiency. The propeller selection charts which were prepared in this research have been found to require extension to a lower range of propeller advance ratio (V/nD), of the order of 0.2, however, in order to be fully applicable to propellers of personal aircraft. This work will be undertaken.

Seaplanes

The Langley Laboratory is continuing its basic research studies on the performance of seaplane hulls, which was described by Mr. Parkinson in his paper. This research includes aerodynamic tests of hull models in a 7- by 10-foot wind tunnel, as well as further work in the NACA towing tanks to study hydrodynamic characteristics. Several such projects are currently in progress and include such items as further studies of the planing-tail hull, hydrodynamic characteristics of planing surfaces and a systematic study of after-body forms of flying-boat hulls.

Power Plants

Several investigations in the field of power plants now in progress in the NACA Laboratories are of interest in personal-aircraft design. One of these projects, a study of new forms of propulsive systems, has just been described in a paper by Mr. Sanders and, therefore, requires no further elaboration on my part.

Interest in safety fuels has been revived since the war, and it has been recommended that research on this problem be carried out to provide a better basis for use of safety fuels in personal-aircraft engines. The Committee's Cleveland Laboratory has recently made engine-performance tests with a safety fuel, and the results of this work showed that the engine performance, with an injection-type impeller in the supercharger, was equal to that of the engine using normal high-test gasoline.

I will presently discuss two further power-plant projects of interest to personal aircraft, these projects being among the several investigations directed toward problems peculiar to personal aircraft.

Icing

Although present types of personal aircraft are for all practical purposes not designed for operation in inclement weather, it is nevertheless necessary to consider that personal aircraft of the future must be capable of both fair- and foul-weather flight in order to realize a satisfactory degree of utility, and, also, of course, safety. The Ames and Cleveland Laboratories at the present time are carrying out a number of investigations on the protection of aircraft against icing. From the results of the work already completed, it can be stated in general that the use of heat in various forms has been found to provide the most effective and dependable means of icing protection for all airplane components, including the wing and tail surfaces, propeller, windshield, radio antennas, instruments, and engine-induction system. It is believed that thermal ice-protection equipment for aircraft can be designed for reasonably low weight as compared to the weights of other anti-icing and de-icing systems.

RESEARCH DIRECTED ESPECIALLY TO PERSONAL-AIRCRAFT PROBLEMS

I now wish to touch briefly upon the several research investigations currently in progress in the Committee's Laboratories which are directed toward solution of problems peculiar to personal aircraft.

Spinning

The first of these projects is an investigation of spinning characteristics of personal aircraft. Mr. Neihouse has illustrated in his paper a criterion of relatively simple application for spin recovery. This criterion, as he mentioned, was based primarily on results of research conducted on military airplanes. Research is being directed at the present time toward verification of this criterion for several typical light airplane configurations, which are expected to include low- and high-wing tractor airplanes, twin-tail boom pusher airplanes, and amphibians.

In addition, these typical light airplane configurations are being utilized in a study of means for making personal aircraft incapable of spinning even if stalled.

Noise Reduction

The second of these projects is the research on noise reduction. As indicated in the papers presented by Mr. Regier and Mr. Vogeley, the Langley Laboratory recently has conducted research to determine the magnitude of the noise propagated by the propeller and the engine exhaust of airplanes and, secondly, to indicate means for reducing propeller noise, inasmuch as the propeller has been found to be the worst offender. This research will be continued, particularly with a view toward pursuing further the more promising solutions to the problem. It is contemplated that this research will include tests on a typical personal airplane to measure the sound levels of the various devices under actual operating conditions. Both engine exhaust and propeller noise will be given attention in this research.

Airfoils

As indicated previously, the Langley low-turbulence tunnel is extending its investigations of the characteristics of airfoils to a low range of Reynolds numbers in order to provide airfoil data which will be directly applicable to the design of personal aircraft wings over the entire speed range of interest. Both the low-drag and conventional types of airfoils are being included in this investigation. The results of this research are also expected to be of value in the interpretation of wind-tunnel tests on airfoils carried out at small scale.

Jet-Ejector Cooling of Personal Aircraft Engines

The fourth project is an investigation being carried out by the Cleveland Laboratory to study the application of the jet-ejector-type cooling system on personal aircraft. In this method of engine cooling the energy present in the exhaust gas is harnessed to induce flow of cooling air through the cowling around the engine. Jet-ejector cooling appears to offer several advantages over other engine-cooling systems, including reduction in cooling drag, design simplicity because of the absence of moving parts, low weight, and possibly some noise dampening. The NACA Laboratories have carried out a certain amount of research on jet-ejector cooling for a large aircraft engine installation, and it is planned to utilize the results of that research in this investigation.

Icing of Light Aircraft Engine-Induction Systems

The last of these projects, and one that is regarded with great importance, is an investigation of means for protection of light aircraft engine-induction systems against icing. Mr. Willson H. Hunter has already indicated in his paper the nature and objectives of this investigation, which is now being organized at the Cleveland Laboratory.